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AVENGER

R.C.N. Avenger AS-3 Mk. 1 with a Wright R-2600-20 engine, as modified from the Grumman TBM-3E by the Fairey Aviation Company of Canada Ltd.

EDITORIAL

PROBLEMS FACING AERONAUTICAL RESEARCH IN CANADA

PRESENT organization, facilities and achievements of aeronautical research in Canada were reviewed by Dr. J. J. Green before the Royal Aeronautical Society in the Eleventh British Commonwealth and Empire Lecture last October. Here, we would like to take a look into the future of aeronautical research conducted by Canadian government agencies, and consider the elements essential to its success.

One of these elements is the existence of an active industry, supported by a stable defence procurement policy. Few countries of comparable population have been so fortunate as Canada in having, since 1945, a continuing policy of development and production.

Another element is the freedom to exchange information with other countries engaged in aeronautical research. In this respect Canada is in the enviable position of enjoying excellent co-operation from the U.S.A. and U.K., and all NATO and Commonwealth countries.

The third element is test and research equipment. Except for two small high speed installations, wind tunnel facilities—the backbone of aerodynamic development—have not been augmented since 1940. For a long time now they have been inadequate to meet the very varied demands, and Canadian industry has been fortunate indeed to be able to carry out high speed tests in American and British facilities. It is expected that around 1960 the situation with respect to high speed equipment will improve, when a 5 ft. wind tunnel will come into operation at Uplands, but one might well ask whether this move should not be paralleled by a similar expansion of low speed facilities. On the structural side too, the need for equipment of larger capacity and size is becoming acute.

With the provision of a large, high speed wind tunnel already under way and the growing need for expansion of other facilities, it is perhaps appropriate to consider whether aeronautical research and test activities, now scattered as small sections of diverse organizations—mostly non-aeronautical and serving both civil and military departments – would not benefit from integration into a single aeronautical framework. Might not such a move, similar to the establishment of Atomic Energy of Canada Ltd., bring better co-ordination of effort, clearer definition of objectives, reduction of administrative ballast and, above all, attraction of badly needed technical talent to the field of aeronautical research?

This brings us to the most important element of all: the people. Clearly, the success of any research organization depends critically on the availability of qualified workers and on the ability of the organization to secure and maintain their services.

The facilities for training in aeronautics have been greatly expanded in Canada during the past few years, with the financial assistance of both government and industry. Although some of us would like to see more emphasis on engineering than on physics in advanced studies, there is no doubt that the framework of aeronautical training is being gradually developed. However, records show that at present the Canadian aeronautical research establishments do not attract enough of the best graduates, many of whom prefer employment in industry and in the U.S.A.

The reasons are obvious enough. The salaries offered by the government appear to be significantly smaller than in industry and particularly in the U.S.A. Furthermore, some workers feel that the pressure of current work in the Canadian establishments does not leave enough opportunity for research. Expansion of research facilities would alleviate the latter complaint. But it seems unlikely that the salary problem can be solved within the current civil service scales and rates of promotion, which were not designed for specialists working in a rapidly expanding and competitive field.

Most governments have had to face this problem. Some have entrusted the operation of their research and test facilities to companies specially organized for that purpose and able to compete with industry for personnel. A solution on these lines might be practicable in Canada. At present, as somebody has recently put it, the government is in competition with itself; it pays industry to lure away many of the men it needs to guide and evaluate what industry is doing.

In this brief review we have touched upon many problems. The future of aeronautical research in Canada will largely depend on their timely solution.

J. LUKASIEWICZ,
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National Research Council.

THRUST AUGMENTATION FOR MILITARY AIRCRAFT†

by Flight Lieutenant W. H. Casley*

Royal Canadian Air Force

SUMMARY

A comparison has been made of the various methods which may be used to augment the thrust of existing military aircraft. Methods which have been considered include uprating the basic engine, afterburning, fluid injection, fitting additional highly rated jet engines, and the use of rocket motors. It is concluded that a rocket motor is a very attractive method of improving the performance of interceptor aircraft at high altitudes for short periods. If longer periods are required, afterburning may well be the most simple method, and is particularly effective at high speeds. The problems inherent in the use of rocket motors are considered to be large, but by no means insuperable, and the military advantages to be gained by the use of such motors are such that their development cannot be ignored.

INTRODUCTION

During the period when piston engines powered the majority of military aircraft, a period extending from the date of the Wright brothers' flights to the middle of the 20th century, aircraft engineers were constantly calling for engines producing more power. In retrospect it is evident that the teams responsible for the design and development of aircraft piston engines might well feel some pride in their achievements, since in less than 50 years of powered flight, absolute power increased from 12 horsepower to over 3,000, and engine specific weight (i.e., lb wt per horsepower) dropped from 14 lb to less than 1. The aerodynamicist and aircraft designer have been quick, however, to take advantage of each improvement in engine performance as it came along, and possibly have occasionally jumped the gun, designing new machines around a promised rather than an achieved performance, so that the engine designer, like Lewis Carrol's famed character, has been running very hard in order to stand still.

The advent of the jet engine seemed to afford some relief, since for a period the engine was capable of propelling the aircraft, in level flight, into a region which had been relatively unexplored, and which soon presented very serious problems to those designing and building aircraft. For a brief period it seemed as though the engine designer could sit back and draw breath, whilst his airframe associates grappled with the baffling problems which arose as the aircraft speed approached the speed of sound in the atmosphere around it. Any respite which may have been envisaged, however, was very short lived, since even if aircraft had, for the moment, surplus power in level flight at some altitudes, more power could be used in military machines to increase the rate of climb, and to provide greater manoeuvreability at high altitudes. Since all fighter aircraft quickly became gas turbine powered, more power was called for to out-climb and out-turn the opposition, and the race continued, and still continues, unchecked.

This paper deals with the methods which might be used to augment the performance of military aircraft which may be considered, under certain operating conditions, to be power limited.

CONDITIONS UNDER WHICH ADDITIONAL PROPULSIVE POWER IS REQUIRED

If an aircraft design is such that no unmanageable change of trim, or unacceptable structural loads occur as the aircraft passes through the transonic region, it may well be argued that such machines will, with existing gas turbine engines, be power limited under certain conditions of flight. The purpose of this paper is, however, to discuss what can be done to assist those less fortunate airframes, which, by virtue of their configuration and aerodynamic limitations, are destined to spend their lives bounded by a stall boundary and a limiting Mach number of less than unity. The majority of such aircraft rely upon a considerable element of brute force, under certain flight conditions, to make them useful fighting machines. Such conditions are:

(a) During take-off with high ambient temperatures, or from high altitude airfields, or both;

(b) During a climb where the existing rate of climb is insufficient to enable the aircraft to fulfil an operational role;

(c) During high altitude combat where additional power is required to increase manoeuvreability, altitude, and speed.

METHODS OF AUGMENTING POWER

Let us assume that it has been decided that a jet-propelled aircraft in military service, or destined for such service, requires additional power to satisfy one or more of the conditions outlined above. The methods which may be used to augment the power are as follows:

(a) Uprate the basic engine(s);

(b) Fit afterburner(s) to the basic engine(s);

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*Directorate of Aircraft Engineering; R.C.A.F. Headquarters, Ottawa.

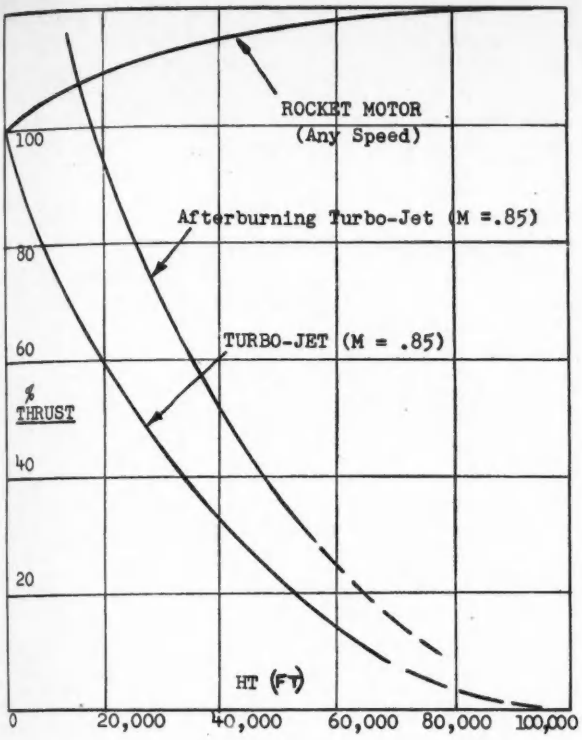

Figure 1
Variation of thrust with altitude

(c) Use fluid injection with the basic engine(s);
(d) Fit additional small gas turbine(s);
(e) Fit rocket motor(s).

A combination of these methods may, of course, be used.

Before considering these methods further, let us look at what happens to the performance of a jet propulsion gas turbine engine under various conditions of temperature and altitude.

EFFECTS OF ALTITUDE

As is common with other "air-breathing" engines the output from a gas turbine engine will fall rapidly with altitude at a fixed forward speed, since reduced density results in a lower mass flow through the engine. At 25,000 ft the thrust is approximately half the sea level value, while at 50,000 ft the engine will produce less than 20% of its rated sea level thrust. Figure 1 illustrates this reduction in performance with altitude and compares the gas turbine with a rocket motor under similar conditions.

EFFECTS OF TEMPERATURE

Figure 2 indicates the effects of temperature on a typical gas turbine. It will be seen that an increase in the ambient temperature to 100°F will cause a reduction in thrust of 10%. Other factors affecting the output from the engine are:

(a) compressor bleed for pressurization and accessories power (about 1% drop in thrust for 1% bleed air),
(b) installation losses due to total head pressure loss in the intake duct and through a debris guard (these losses may well be 5% of the rated thrust).

Figure 2 also indicates the approximate installed performance of an engine operating from a 5000 ft altitude airfield with an ambient temperature of 100°F. When the aircraft has reached its normal take-off speed the thrust of the engine has dropped to 60% of its maximum Model Specification rating. Some form of augmentation is necessary if the aircraft is to operate with normal load under such conditions, to prevent take-off distances from becoming prohibitive.

UPRATING THE BASIC ENGINE

During the evolution of a gas turbine it is customary for an increase in power of 15-25% during its production life. This is achieved by the following means:

(a) Increasing component efficiencies,
(b) Increasing the mass flow without changing the envelope size of the engine,
(c) Increasing the cycle temperatures.

This normally leaves the weight unchanged, and is the sort of bonus which an aircraft designer expects. The Orenda, for example, has increased its output by about 25%, while decreasing its specific fuel consumption considerably, and its weight by 11%. It can be readily appreciated, however, that the change in total thrust at high altitudes is very small.

AFTERBURNING

The output from a conventional gas turbine is limited by the maximum cycle temperature tolerable to the turbine blades. Turbine entry temperatures of the order of 900°C (1600°F) are used in engines in service today, and, after expansion through the turbine section, the gas temperature is of the order of 650°C (1150°F) in the engine tail pipe. Afterburning is a process of burning

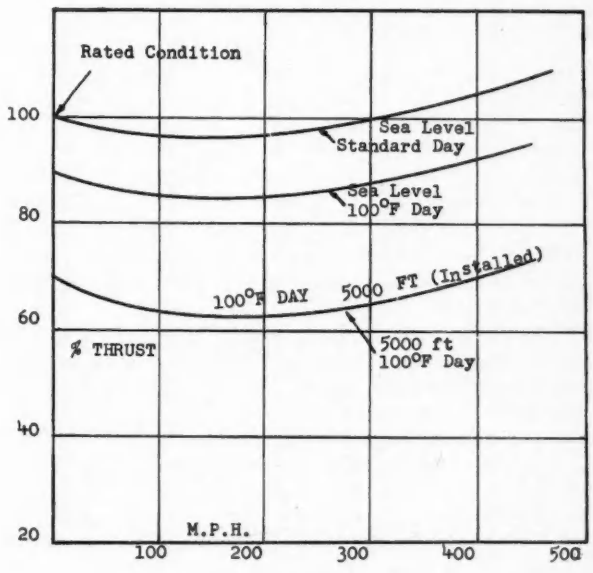

Figure 2
Variation of jet engine thrust with ambient conditions

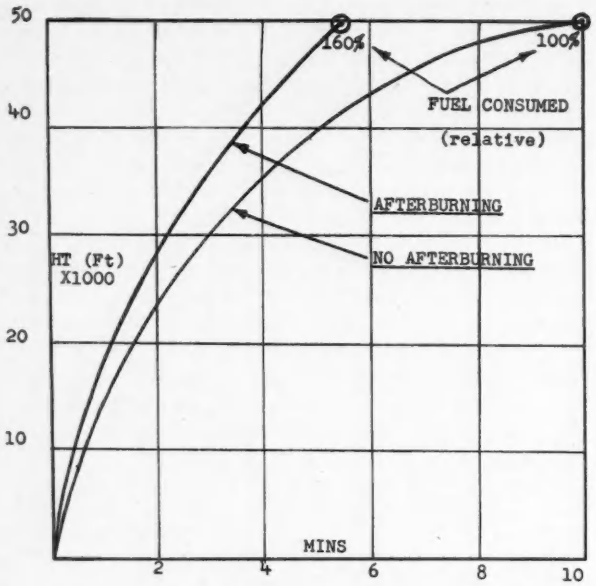

Figure 3
Effect of afterburning on climbing performance

fuel in the tail pipe to increase the energy of the gas after it has passed through the turbine. Since the gas is expanded when additional heat is added in the tail pipe, the final nozzle area must be increased if conditions at the turbine are to remain unchanged, and this gives rise to one of the most serious problems associated with an afterburner, as the engine demands one size of final nozzle when operating normally, and a larger size when the afterburner is ignited. This calls for a variable area nozzle which must operate in a gas stream which will readily melt mild steel, presenting a formidable design and development problem.

By means of an afterburner, the static thrust of a gas turbine may be increased as much as 50%, but this is accomplished at the expense of fuel consumption. The specific fuel consumption (i.e. lb of fuel consumed per lb of thrust per hour) of a non-afterburning engine is normally near 1.0, while with reheat it is near 2.0, so that the specific fuel consumption per lb of static *afterburner* thrust is of the order of 4.0 lb/lb/hr. However, as a climbing aid the afterburner is not quite as unattractive as this may at first sight lead one to believe, since time to a given height is greatly reduced and the overall fuel consumed in the climb is not necessarily much greater than would be the case when climbing without afterburning. (See Figure 3)

One of the attractive features of the afterburner is the manner in which the percentage augmentation increases with forward speed. To assist in understanding this, the following explanation in simple terms is offered. The net thrust produced by a jet engine may be defined as the thrust developed within the engine less the initial momentum of the air taken into the engine. The afterburner is used to augment the thrust developed within the engine, (gross thrust since it is unaffected by the intake conditions), so that at high forward speeds where much more air is being swallowed by the engine the percentage augmentation increases. (See Figure 4) It will be noted

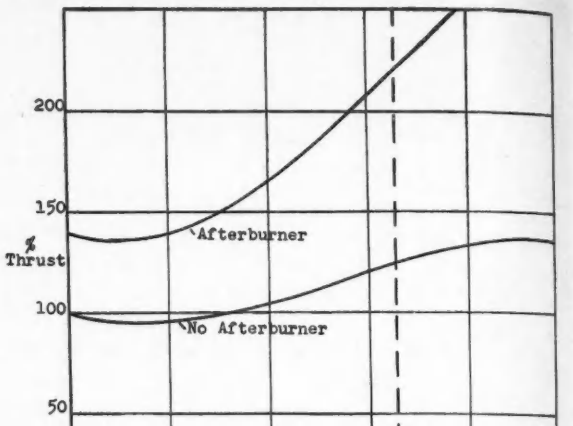

Figure 4
Thrust vs Forward Speed at 50,000 ft.

that an afterburner giving 40% boost under static conditions is producing 75% at M = .85. If this curve is projected into supersonic flight conditions, it will be seen that the afterburner produces far more thrust than the basic engine, and becomes an essential adjunct. It is basically a ram jet operating behind a jet engine, and thus becomes a ram jet capable of providing static thrust.

An afterburner such as described above would normally weigh 20-25% of the basic dry engine weight, but there are quite extensive airframe changes required to fit such a device, since cooling problems are severe, and the tail pipe diameter is increased, becoming as great as that of the basic engine if high augmentation ratios are required. (See Figure 5)

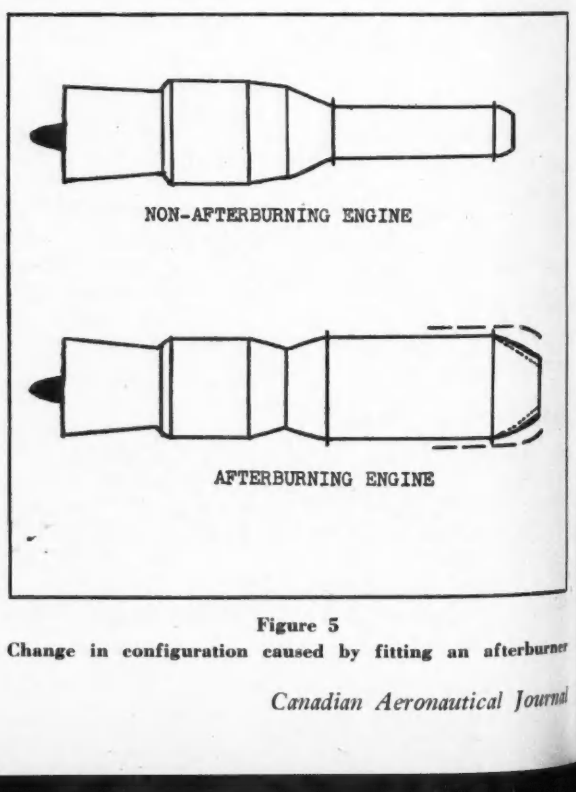

Figure 5
Change in configuration caused by fitting an afterburner

FLUID INJECTION

Afterburning, as described above, is a means of increasing the momentum of the gas leaving the engine, by increasing jet velocity. Fluid injection can be used to accomplish the same end by increasing the mass flow. If water or a water/alcohol mixture is injected into the compressor, it lowers the temperature of the charge entering the engine, increasing its density, and increasing the mass flow. Very large quantities are required, however, to give a substantial thrust increase, and for this reason this type of augmentation is best used for restoring take-off power when operating from high altitudes or tropical airfields. For example, to restore a 10,000 lb engine to its original rating when operating on a tropical summer day, when its output will have dropped 1000 lb, would require an injection rate of approximately 1 gallon per second into the compressor. The advantages offered are as follows:

(a) Having used the fluid, the tanks may be jettisoned, and the operating characteristics of the engine remain unchanged. (An afterburner tail pipe gives rise to a 2-4% loss in cruise thrust, and a similar rise in specific fuel consumption.)

(b) There is no basic engine weight penalty worthy of note.

FIT SMALL GAS TURBINES

Highly rated, short life jet engines are already in existence, and may be used to augment the power of the basic power units. Such engines would normally be pod-mounted, and with fighter aircraft, one would expect one to be fitted at each wing tip, unless such obvious drawbacks as large sweep-back prevent this. These units suffer from the same deficiencies as all gas turbines, their thrust decreasing severely at high altitude where augmentation is most needed, but an engine designed for high output and short life may be very light, and a 5,000 lb engine should weigh no more than 1,000-1,200 lb (installed wt). If such engines are to be used for high altitude augmentation, it must be possible to start and accelerate the engine under these conditions, and this may pose new problems, since the existing engines of this type may not have been designed with this type of operation in mind.

The great advantage favouring the use of gas turbine engines in this role, lies in the low fuel consumption of such units, so that for lengthy periods of augmentation such a system appears attractive. The drag and weight of the units, however, result in a marked deterioration of the aircraft performance when operating without augmented power.

FIT ROCKET MOTORS

The liquid rocket motor has one very great advantage over any other form of augmenting device, and one equally large disadvantage. Its great advantage lies in the fact that its thrust is almost independent of altitude (it increases slightly), so that a 2,000 lb thrust rocket motor will produce more thrust than a 12,000 lb thrust gas turbine when both are operating at 50,000 ft. Its ability to thus perform, however, provides the clue to the rocket motor's greatest drawback, since it must carry its own oxidant, while the gas turbine draws oxygen from the surrounding atmosphere. When a rocket motor is fitted to an aircraft, therefore, arrangements must be made to store and pump large quantities of both fuel and oxidant, supplying these to a combustion chamber at the correct pressures and in the correct ratios. This requirement also presents additional problems to ground servicing crews who may be called upon to handle such difficult fluids as fuming nitric acid, high test hydrogen peroxide, or liquid oxygen.

Since the rocket motor carries its own oxidant, its total propellant consumption is enormous, and rocket motors are, of necessity, short duration devices. As an example, a 2,000 lb thrust motor will consume 18 tons of propellants per hour, while a jet engine of similar power consumes 1 ton. However, it will be shown later that the rocket motor should not be discounted if short augmentation periods at high altitude are required, since the motor itself is very light (only about 5% of the weight of a gas turbine giving similar thrust at 50,000 ft). Some of the problems associated with the use of rocket motors for thrust augmenting are as follows:

(a) Hazards

While a perfectly functioning rocket motor offers no greater hazard than any other form of engine, any malfunction is likely to be more serious since very large quantities of explosive mixture are involved. For example, failure to ignite at precisely the correct milli-second results, in the case of the 2,000 lb motor mentioned above, in 10 lb/sec of propellants flowing through the combustion chamber with disastrous results when ignition takes place.

(b) Storage of Propellants

There are various combinations of fuel and oxidants which may be used in rocket motors. The author considers, however, that the most suitable combination at the present time is that of hydrogen peroxide and gas turbine fuel. There are two main advantages favouring such a combination, in that the fuel is that already in use in the aircraft fuel system, while hydrogen peroxide can, by decomposition, provide energy to drive turbo pumps to deliver the large mass of propellants to the combustion chamber, making the rocket motor self-contained. A drive shaft from the gas turbine accessories gear box provides an alternative method of pumping propellants to the rocket motor. It should be remembered, however, that the objective is to fly the aircraft under combat conditions at very high altitudes where the stability limits of the gas turbine are small, so that manoeuvres made possible by the provision of rocket power may result in gas turbine blow-out, leaving the aircraft with no means of propulsion. The simplicity of a mechanical drive is highly desirable, but the operational consideration outlined above may override this great advantage. Other propellant combinations include:

(i) liquid oxygen/hydro-carbon fuel;

(ii) nitric acid/aniline;

(iii) low concentration hydrogen peroxide as a monopropellant, with low combustion chamber temperatures, and a specific impulse about 50% that of the combinations mentioned above.

Liquid oxygen has an extremely low boiling point (-180°C), and hence evaporation losses are very high so that oxidant tanks must be constantly replenished if a fighter spends long periods at standby. The main drawback to nitric acid lies in the highly corrosive properties of the acid, making the design of the rocket systems difficult, and the handling problems acute. High test hydrogen peroxide (HTP) is easier to handle in these respects and also, when decomposed in the presence of a suitable catalyst, reaches a temperature which will ignite the fuel without the assistance of any other form of ignition system.

(c) Control of Output

When a rocket motor is used to augment the power of a fighting aircraft at high altitude, the question of what degree of rocket thrust control is required is one which could result in many hours of controversial discussion among pilots. To simplify the control system, the author offers the following proposal.

In the case of an interceptor, the additional power is required either to increase altitude rapidly, or to get into position for an attack. In either case the pilot requires all available power, so that an "on-off" type of control by means of a button on the control column or throttle lever may be sufficient. Reduction in total thrust can readily be made by reducing the output from the jet engine(s), while the rocket output may be terminated at any time by pushing the power control button. During an attack, however, it is essential that terminating rocket power should not affect the aircraft trim in such a manner as to render aiming difficult.

COMPARISON OF VARIOUS METHODS OF AUGMENTATION

Figure 6 and Table 1 have been prepared to compare the various types of thrust augmenting schemes which have already been discussed. It has been assumed that 2,000 lb of additional thrust are required at 50,000 ft to enable an interceptor aircraft to perform its mission. The following means provide the additional power:

(a) Two gas turbines, each rated at 5,000 lb sea-level static thrust, and each weighing 1,100 lb installed;

(b) A 2,000 lb thrust rocket motor, weighing 250 lb installed, and using propellants having a specific impulse of 200 lb/lb/sec;

(c) Afterburner(s) fitted to the basic engine(s). If the basic power units are 6,000-7,000 lb thrust gas turbines, afterburning to about 1700°K would provide an additional 1,000 lb from each engine at a Mach No of .8 (approx) at 50,000 ft, at an sfc per lb of afterburner thrust of approximately 4.0.

It will be noted that for a period of less than 3 minutes, the rocket motor is the lightest form of thrust booster, while afterburning is the most attractive if between 3 and 8 minutes is required. The highly rated gas turbine, because of its low consumption, is the lightest system if lengthy periods of thrust augmentation are required.

It is the author's belief that the rocket motor may be more simple to install on existing fighter aircraft than either of the other systems, since it is smaller and lighter, and normally need not involve the major structural changes required when either afterburners or additional jet engines are fitted. Such augmentation may well mean the difference between a successful interception and a useless mission, while the aircraft, when operating with empty propellant tanks, loses virtually none of its original performance. The problems involved in introducing such a novel propulsive system are great, and considerable training is essential if the hazards are to be reduced to a reasonable level, but the operational advantages are such that they cannot be ignored.

TABLE 1
COMPARISON OF THRUST AUGMENTATION METHODS

Type of Augmentation	% Boost Possible	Fuel Cons per 1000 lb/hr	Remarks
Basic Engine Uprating	Normally 15–25%	1000 lb	Advantageous for all Flight Conditions
Afterburning	50% Static 100% –M = 1.0	4000 lb	Particularly Good at High Speed
Fluid Injection	Normally Limited to 10%	36,000 lb (Water)	Useful for Take-off
Small Jet Engines	As Required	1000–1200 lb	Advantageous for all Flight Conditions
Rocket Motors	As Required	18,000 lb	Particularly Useful at High Altitude

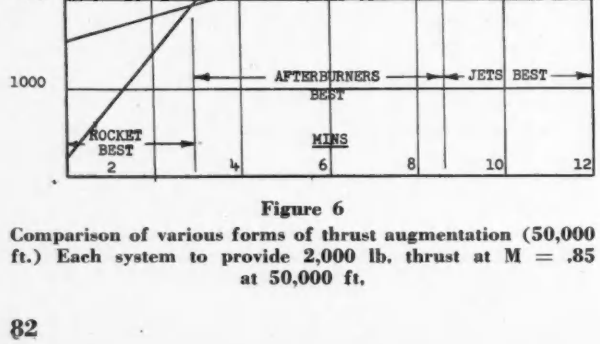

Figure 6
Comparison of various forms of thrust augmentation (50,000 ft.) Each system to provide 2,000 lb. thrust at M = .85 at 50,000 ft.

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THE PROBLEM OF THE VERY LIGHT-WEIGHT, HIGHLY-EFFICIENT AEROPLANE†

by **B. S. Shenstone***

British European Airways

INTRODUCTION

THE purpose of this paper is to discuss the problem of the minimum power needed to enable a man to be flown. It is intended to be a survey of the efforts that have been made over the last few years on this problem, and to bring to your attention particularly the latest thoughts and accomplishments in this direction.

We all know what power is required to fly fifty people at various speeds which, in many cases, are quite efficient speeds, because people can afford to pay for them. On the other hand, are we convinced that they are the most economical conditions? I think the quick answer to that is that they are not economical, but only the most competitive conditions. Only in the early days of flying was attention given to flight with a minimum power, because in those days there was no more than minimum power available in the engines that could be used. Once engines had become sufficiently powerful to fly one or more men about the sky, this problem was lost sight of and has very seldom been discussed since.

It is of interest to know that on December 17, 1903, 12 hp was the minimum with which a man could fly. In fifty odd years this power has become less, but more is always used.

In explaining the assumptions I am making (for the solution of all problems involves the establishment of assumptions), I would say that I shall investigate the economics of the problem only from the point of view of economy of power. I shall not be concerned whether the airframe or the source of power is extremely expensive, but only devote my efforts to the one point—that of the minimum power to fly a man.

Sir George Cayley over a hundred years ago gave a great deal of consideration to this problem, as did Lilienthal, the Wright Brothers and many others. It should be noted—but not with scorn—that both Lilienthal and Cayley made mistakes in their assumptions and misled themselves on this problem. It is therefore wise for my assumptions to be suspect, because it is not only the famous that mislead themselves.

Who wants light, highly efficient aircraft?

1. Those who think the light aeroplane needing more than 50 hp is no longer light. If a sharp improvement in economy were possible, maybe a really light aeroplane would be usable.
2. Those who would like to have some auxiliary power on their sailplanes, to extend their usefulness.
3. Those who want a very light efficient glider that can be launched by the pilot.
4. Those who want to fly by manpower.

WEIGHT AND DRAG

The general problem is one of weight and drag. Our payload, the man of standard weight of 165 lb, is constant. Everything else is fluid, that is, the materials, the shape, the speed and anything else you think of. Lilienthal concentrated on a light-weight structure made of cloth and osiers. This arrangement weighed approximately 65 lb, so that the all-up-weight was 65 + 165 lb = 230 lb, and his gliding angle was approximately 5.5/1, which indicates that the power needed to fly level would have been about 2 hp at 16 mph.[a] Figure 1 shows a reproduction of one of Lilienthal's gliders. Here we have a man being flown under the maximum natural drag conditions, hanging vertically and unfaired. Admittedly we could increase the drag if we tried, but it is even easier to decrease it by adding fairings to the structure and thereby making the whole thing heavier. The point here is as follows: Can we reduce the total power required in spite of adding weight? The answer to this is in general "Yes", but of course there is a limit and that is what we must discover. As in all design work, the solution comes out of balancing incompatibles one with another, and getting out something like an optimum solution.

†This paper was presented on the 12th November 1955 to the Low Speed Aerodynamics Research Association in London, England. A preliminary version of it was read by Mr. Shenstone before the Winnipeg and Toronto Branches of the C.A.I. on the 27th and 29th September 1955. It is published by permission of the Low Speed Aerodynamics Research Association.

*Chief Engineer.

[a]Information given in Lilienthal's book "Der Vogelflug als Grundlage der Fliegekunst" on his gliders is quite insufficient for the calculation of the gliding angle or any other aerodynamic characteristic. One can take a shot at the aerofoil drag, but the parasitic drag and interference are immense and unknown. Even the number of wires and their diameter were not given. All gliding angles quoted are against winds of varying strength in a vertical direction. This figure of 2 hp is therefore suspect, but nevertheless is not likely to be so wrong as to vitiate the remarks which follow.

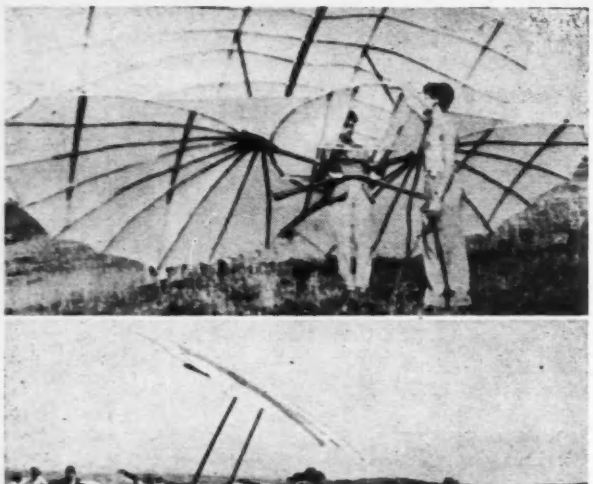

Figure 1
Reproduction of a Lilienthal glider.

Vertical elevation

However, before getting down to a certain amount of detail, it may be worth while considering the simple case of the power required to elevate a man vertically above the ground without forward motion. Many people have been captivated by the thought of a tiny light helicopter to enable a man to equip himself with approximately seven-league boots. In this simple problem of elevation, there is no problem of drag. But the power required to lift the man one inch or one foot or any small altitude is approximately 12 hp, assuming a 10-ft rotor. Since a mechanism must weigh something—and it would not be pessimistic to say that any sort of motorized rotor weighs 100 lb—the total weight to be elevated would be 265 lb at 19 hp. That is clearly not the way to fly with minimum power.

AERODYNAMIC REFINEMENTS

First, we shall see what can be done about the aerodynamics of the job, and later we shall look at the structural problems and what sort of drive would be practicable.

I shall try to avoid theory and use available facts as much as possible, sometimes twisting them into the right shape. So I start almost twenty years ago, when there was a contest in Germany for aircraft operated by man-power. This inspired efforts in various countries, and four of these efforts are worth mention although none of these machines was successful. They are worth mention because they achieved certain low weights and good aerodynamics which will give us another signpost on our way and avoid a certain amount of guess-work.

These aircraft were the Bossi-Bonomi (Italy), the Haessler-Villinger (Germany), the Seehase (Germany) and an un-named Russian. Data on and powers required for these aircraft are shown in Table 1, and it will be seen that in spite of the greater weights compared with Lilienthal's all-up-weight, their powers required are considerably less. In fact, they are far less than that required for even a small high efficiency sailplane. (Windspiel 1.3 hp; Tiny Mite 2 hp).

Figure 2 shows the general arrangement of the Bossi-Bonomi. Figure 3 is a photo of it. Figure 4 shows the general arrangement of the Haessler-Villinger and Figure 5 a photo of it.

Of these, the Haessler-Villinger is the most attractive. It has a low weight, a low power requirement and an

TABLE 1
DATA ON FOUR MAN-POWERED AIRCRAFT.

	Haessler Villinger	Bossi-Bonomi	Seehase	Russian
Span ft.	44.3	55.7	42.7	39.4
Wing Area sq. ft.	108	226	160	149
Aspect Ratio	18.25	13.4	11.4	10.4
Weights lb.				
Wing		99	40.7	59
Fuselage	77	79.2		48.2
Tail			38.3	5.7
Equipment	24.2	35.8		9.1
Empty Weight	101.2	214	79	122
Load	143	143	143	143
All-Up Wt.	244.2	357	222	265
Ultimate Factor	6	4 ?	4 ?	–
Fuselage Cross Section Area sq. ft.	3.55	–	–	3.74
Wing Root Section	G 535	0012-F1	G 497	–
Best L/D.	24	27.5	–	22
Best Sink Speed fps.	1.7	1.05	–	1.8
Wing Loading lb./ft.^{2}	2.26	1.58	1.39	1.78
Span^2 Loading lb./ft.^{2}	0.12	0.114	0.12	0.17
Pedal rps.	–	1	–	1.4
Propeller rpm.	500	150	> 300	500
Propeller Diam. ft.	4.93	7.4	9.4	3.9

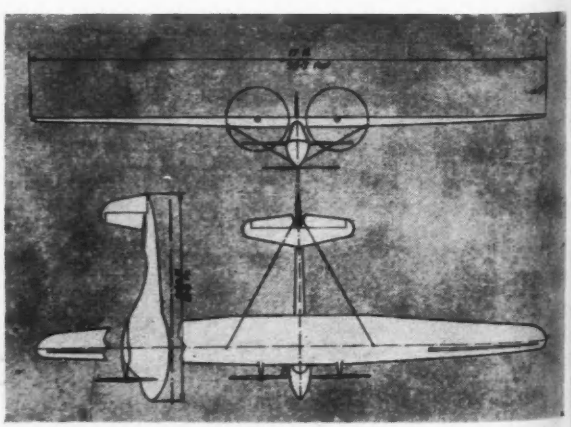

Figure 2
General arrangement, Bossi-Bonomi.

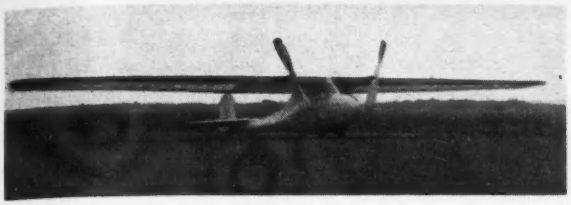

Figure 3
Photo—Bossi-Bonomi

efficient propeller position. The Haessler-Villinger will therefore be used as a basis for the following variations.

Raspet's work

Now what has been accomplished in the last twenty years? The answer is fairly simple: very little has been done. What has been done is indirect work by Raspet on the reduction in drag of sailplanes, which has resulted in much higher performance than had been expected hitherto. Raspet's technique is extremely simple and is

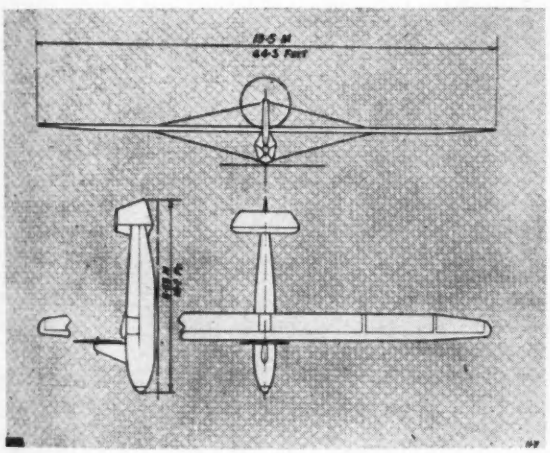

Figure 4
General arrangement, Haessler-Villinger.

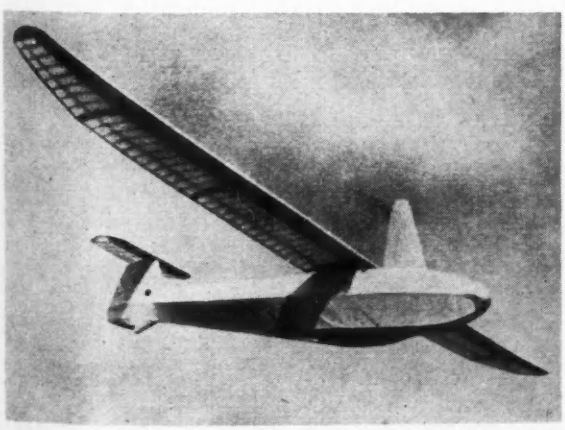

Figure 5
Photo—Haessler-Villinger

concerned with doing what we all know can be done but never do enough of, that is:

1. Make the surfaces as smooth as possible.
2. Get rid of leaks.
3. Eliminate interference between surfaces.

These points are so obvious that they should need no explanation. The difficulty is purely one of degree. As smooth as possible must be defined as to waviness and detail roughness, and these must be measured and the standard laid must be adhered to. This is difficult and laborious. To get rid of leaks is also no easy matter and requires the most careful survey of all possible apertures into wing and fuselage. The elimination of interference can only be done experimentally after all obvious knobs, such as hooks, pitot heads, wheels, skids, control-horns, gaps, etc. have been eliminated. The elimination of many of these obvious things may mean re-design and often it is too much trouble. You can realize this if you examine any so-called clean aeroplane. But in this case everything counts and everything must be done, and it may take months to do.

The work can be summarized by saying that Raspet has been able to show that a minimum drag coefficient of 0.01 and high span efficiency factors are attainable, which can be applied to the aircraft type we are considering. Raspet's work was done on a 40.3-ft span sailplane with the ridiculous name of Tiny Mite (shown on Figure 6).

At 30 mph the Haessler-Villinger (77 lb empty weight) required only 0.82 hp. If the improved drag coefficient were applied to an aircraft weighing the same as the Haessler-Villinger, it would be seen that the minimum power required is reduced from 0.82 hp to 0.68 hp.

Ground effect

If we not only clean up this machine but fly it close to the ground (say 10 ft) the power required is reduced to 0.58 at 30 mph because of the reduction of induced drag caused by ground effect.

Boundary layer control

So far, only the most straightforward and simple means for reducing drag have been suggested. Even so, the estimated power required has dropped to 70% of the originally achieved condition. Let us now turn to the

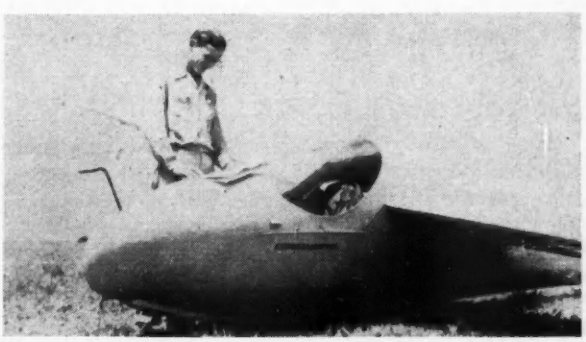

Figure 6
Tiny Mite.

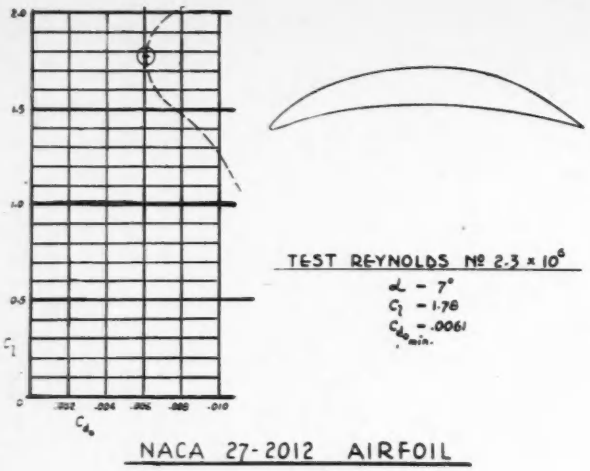

Figure 7
Wing Section NACA 27-2012

pilot and ask him to exert himself a little, say at the rate of 0.1 hp, which he can do indefinitely without tiring. This expenditure of power applied to boundary layer suction should enable the whole wing to achieve laminar boundary layer conditions and reduce the power required for flight by about 0.2 hp, which is pretty good for an expenditure of only 0.1 hp. With ground effect, this could reduce the power required to about 0.38 hp.

Aerofoil section

In these estimates I have simply accepted the Tiny Mite aerodynamics, particularly the minimum drag coefficient, which occurs at a lift coefficient much lower than those we are concerned with. The section used (RAF-34) is a low cambered reflex section which was in use for aeroplanes over twenty years ago. It is probable that higher cambered sections designed for high angle of attack flying would give better results than RAF-34. In this connection, R. D. Hiscocks has suggested that perhaps NACA 27-2012 would be worth trying. This has 14% camber at half chord and maximum thickness at 60% chord, a veritable skyhook of a section having a high incidence laminar bucket. (Figure 7)

In Figure 8 the following polars are shown, using the square of the lift coefficient for convenience:

1. Original Haessler-Villinger
2. Tiny Mite
3. Improved Haessler-Villinger (not including suction).

STRUCTURAL REFINEMENTS

That is what may be possible with aerodynamic improvements alone. We must now consider the structure and see what can be done. The development of sailplanes and power aircraft of moderate and small sizes has not led to particularly low structure weights. The sailplane is not sensitive to weight and in fact quite a heavy weight is sometimes an advantage in cross-country flying. The light-weight glider is not particularly attractive because very light structures tend to be expensive not only in design effort but in manufacture. Hence, there remains a great deal of scope for developing light, small-scale structures. Even the ply-wood and spruce fabric-covered wing can be made much lighter than it usually is. In almost all cases the rib weights are several times greater than necessary, not for air loads but for ground loads. The very scale of these low-powered aircraft reveals new structural problems which in some respects are akin to those met in large flying models. The low weights actually achieved may surprise some of you, but the flying conditions are so different that very light structures are practicable. The speeds are low, the manoeuvre loads are low and the load factors may be as low as 4 if you wish.

By ignoring existing types of structure, there is a great deal to be done in producing structures from the outside in; that is, working from the desired aerodynamic shape inwards to the structure rather than wrapping the shape around the structure. There is much scope here for light sandwich structures, plastics and even light metals such as magnesium. A mixed structure might be better than a single-material structure, but this is always difficult to produce because people tend to be enthusiastic about one sort of structure and scornful of all others. There is no doubt that a pure metal structure of the type now used on larger aircraft would be quite out of place because the scantlings would be so light as to defy riveting and there would be tremendous trouble with detail structural instability and minimum gauges. Any metal structure would have to be supported by some light, stiff material, and it would be extremely difficult for me to say anthing definite about the advantages of one type of structure over another.

Seehase's design

I think it is worth noting the work done by Seehase about seventeen years ago. Seehase had been doing research on man-carrying kites for many years, and applied his special knowledge to man-powered aircraft. He based his wing, which had a span of 13 metres (43 ft), on the use of only four ribs per side. These ribs were attached to two magnesium tubes which acted as spars and over everything was stretched doped silk. The only

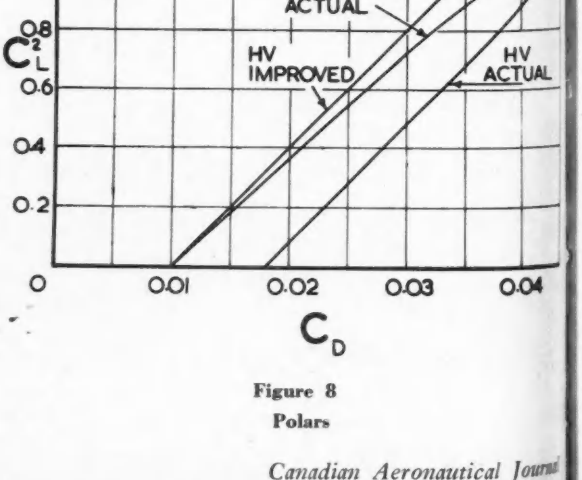

Figure 8
Polars

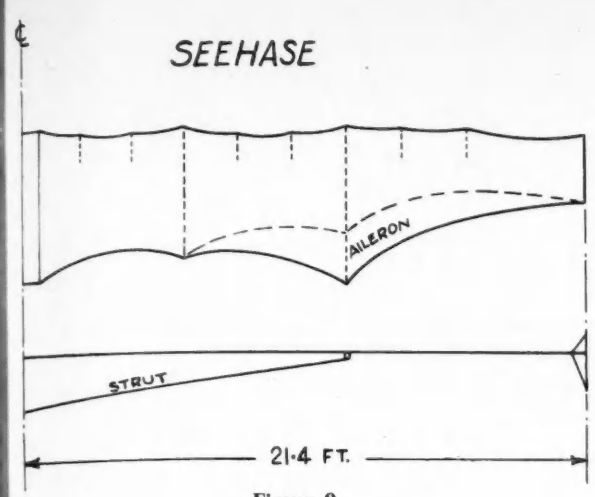

Figure 9
Seehase's wing

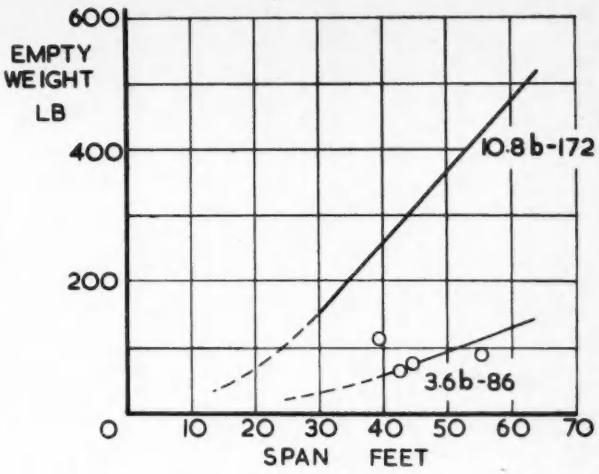

Figure 10
Empty weight vs. span

other structures were a pair of nose ribs between each of the main ribs. Figure 9 is a sketch of this wing. The whole machine, complete with control surfaces and machinery for driving it by manpower, weighed 36 kg (79 lb), making a gross weight of 244 lb.

I am not saying that Seehase's aircraft was aerodynamically as clean as it should be, but it does show to what extent the structure weight could be reduced by ingenuity.

WEIGHT AND AERODYNAMIC CONSIDERATIONS

I think the main point is that the structural problem is probably far more difficult than the aerodynamic problem, because it needs more patience.

For a given speed the power required is directly proportional to weight. However, most of the weight, that of the pilot, is out of our control. The sort of weights we are considering are structures of the order of 75 lb and of course our man at 165 lb, making a gross weight of the order of 240 lb only about one-third of it being under control. No matter what we do it is unlikely that we could as much as halve this, making a gross weight of 203 lb, lowering the power required by 15% or about 0.06 hp. To be realistic, there is not really a great deal to be accomplished by decreasing the weight much below that already achieved, but there may be much to be gained by improving aerodynamics for that same weight.

It is most important to look at the weight/aerodynamics problem. Although the power required is directly proportional to the weight at a given speed, it is also inversely proportional to L/D. If we can improve the L/D without a proportional weight increase, we are gaining. How can we best improve the L/D? If we look at the polar of the Haessler-Villinger cleaned up to Tiny Mite standard, we find that at the low speeds we are considering (C_L about 1.0) the total drag coefficient is about 0.03, and of that the induced drag coefficient is 0.0187, which is 62.3% of the total. This is worth looking at. To reduce the induced drag we must increase the wing span and this means adding weight. At this point it might be of interest to make an elaborate series of calculations on the effect of span on wing weight. Because such calculations can easily lead one astray and have in the past led me astray, I shall again fall back on the few facts available and see what comes of them.

Data on a large number of sailplanes indicate that the empty weight varies linearly with span over a range of 30 ft to 60 ft. The effect of aspect ratio is very secondary. In Figure 10 is shown the line $(10.8b - 172)$ which is the empty weight in lb when b is span in feet. Also shown are four aircraft designed and built for manpower, but omitting (as far as detailed weights or lack of them allow) the mechanisms and propellers. It is to be expected that these aircraft would be lighter than sailplanes, and so they are. Could we draw a curve through their weights parallel to $(10.8b - 172)$? Hardly, for that would need an almighty kink ever to reach zero at zero span. So the slope must be less and I suggest a curve $(3.6b - 86)$ as reasonable. The advantage of this sort of approximation is that the degree of arbitrariness is clear and alternatives can easily be devised with a minimum of mental exertion by the audience while I am speaking.

Very well, what if we increased the span of the Haessler-Villinger from 44.3 ft to 60 ft? The gross weight will be increased 53 lb, i.e. to 297.2 lb. As for wing area, let us take two alternatives, which are shown in Table 2:

TABLE 2
DATA ON TWO 60 FT SPAN AIRCRAFT.

	Original Wing Area	Original Wing Loading
Span ft.	60	60
Wing Area ft.²	108	131.5
Aspect Ratio	33.2	27.5
Wing Loading lb./sq. ft.	2.75	2.26

In the case when the wing area is not increased, calculations show about the same power required, but if the wing loading is kept constant, there is a slight improvement of about 0.05 hp. But if laminar flow can be achieved and ground effect used, the power required may come as low as 0.35 hp + power for laminarization.

This does show how a proper use of weight can result in lower power required. Thus, over-generalizing, we say that by making an aeroplane 30% heavier, we have cut down the power required by nearly ⅓.

MANPOWER

This means that even a man could fly such a machine. Not for long, but real honest flying. Assuming him launched at about 30 mph, and an efficiency for the drive and propeller of 75%, he would have to exert

$$\frac{0.35 + 0.1}{175} = 0.6 \text{ hp.}$$

With feet alone he could fly 50 sec or ¼ mile.

With hands and feet—indefinitely.

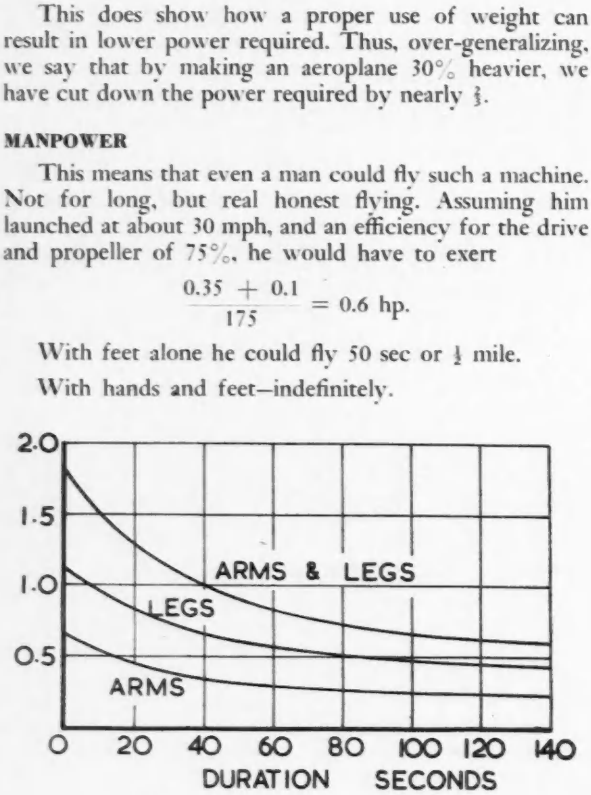

Figure 11
Manpower available and variation with time

The data on which this is based were collected by Oscar Ursinus and published in 1936 and 1937 in "Flugsport". The experiments covered all likely methods producing manpower and took into account the attitude of the man, different sorts of motion, different speeds of motion and the use of hands and feet separately and together. Figure 11 summarizes the amount of power available with time under optimum conditions which consist of pedalling motions. The best speeds for use over long periods are about 1 rps for arms and very slightly faster for legs. Rowing motions were found to be inefficient. But nobody could use both hands and feet for propulsion and control the aircraft as well. Let us not ask the impossible. If a two-seater aircraft were developed for manpower, it could have a better power for its weight. From Wilkinson's work it is clear that for a given span and aspect ratio, the two-seater glider need have an empty weight not more than 20 lb greater than the single-seater.

Thus, if we put two men in the ultimate aeroplane described above, we would increase the weight by 20 + 165 lb = 185 lb, making the gross weight 300 + 185 = 485 lb. Taking the same lower wing loading as used for the Haessler-Villinger of 2.26 and 60 ft span, we find that

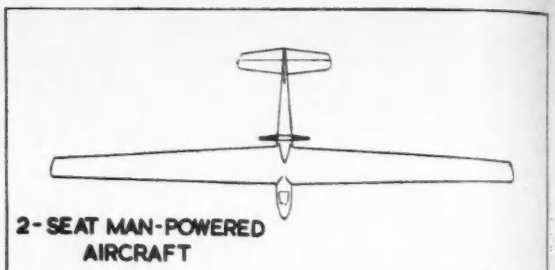

Figure 12
General arrangement of two-seater

an all-out laminar plus ground effect assumption requires 0.5 hp. One man, the galley-slave, exerting both legs and arms and the pilot working only his legs, can produce a power of about 1 hp for an indefinite period if in training. If we assume that suction requires 0.1 hp, the propulsive efficiency can be as low as 60%.

It is an obvious step from the skiff to the trireme, and there is no reason why a multi-manpowered aircraft should not be built. It could be built now.

What might such a two-seater look like? I haven't designed one, but a three-view sketch often gives the impression that something has been designed. In fact, it is possible to get orders for aircraft on the strength of a three-view. Figure 12 shows what might be done if Dr. Raspet were in league with Mr. Haessler and Mr. Villinger. The propeller not only drives, but through its hollow blades, pumps out air from the wings to provide the laminar boundary layer. Whether such a pump is properly efficient I don't know, but it is an idea, although not mine. Launching would be by towing or catapult. One should be careful to take off and land with the propeller stopped and blades horizontal.

MEANS OF PROPULSION

The third problem is that of the means of propulsion and this has had far too little attention paid to it in the past. The idea of attaching small motor cycle engines with propellers has been carried out again and again and again, but with very little success because of the excessive drag of the unit.

It is realised that the L.S.A.R.A. and other organizations have tried to get some engines developed, but so far any engines flying are of very excessive power or, if their power is low enough, they are not yet dependable. In most cases where these small engines have been applied, the aircraft itself has not been very efficient aerodynamically or structurally. There seems to be no reason why even auxiliary pedal-cycle engines should not be used to drive an aircraft. However, the utmost effort would have to be made on the aerodynamic and structural sides to make it practicable, but if it does become practicable, the relative weight of the power unit compared to that of the

payload would be greatly improved over any flown to date. The problem of efficiently cooling a very small single or twin-cylinder engine has never been tackled. It is no job for an amateur, but somehow it would have to be done if a truly efficient aeroplane is to be produced. However, the obvious thing is to use the payload to drive the aircraft, that is to say, use the power of the man rather than let him sit there eating his head off or biting his nails. But if we don't want to use him, we could add an engine which could be as small as 3 hp to fly one man at 50 mph.

Before approaching details, it may be worth while to discuss the actual means of propulsion. Only three possibilities occur to the speaker:

Jet propulsion
Propeller propulsion
Flapping wing propulsion

As far as the jet is concerned, the speed of progression of a low-powered aircraft is so slow that a jet would be highly inefficient, so we must consider the good old propeller, in some form or another. It is fortunate that so much is known about the characteristics of propellers, and we can therefore deal with it with some confidence. However, we must remember that in the aircraft under consideration, the aerodynamics are very sensitive to disturbance and therefore it would be most unwise to use tractor propellers. Hence, we must assume the use of a pusher which does not affect the flow over the wing. If a tail is used in the design in the normal position, it would be wise to assume that the flow over the tail has a turbulent boundary layer. Therefore, we can limit ourselves in finding a method of driving the propeller which is as light and as efficient as possible.

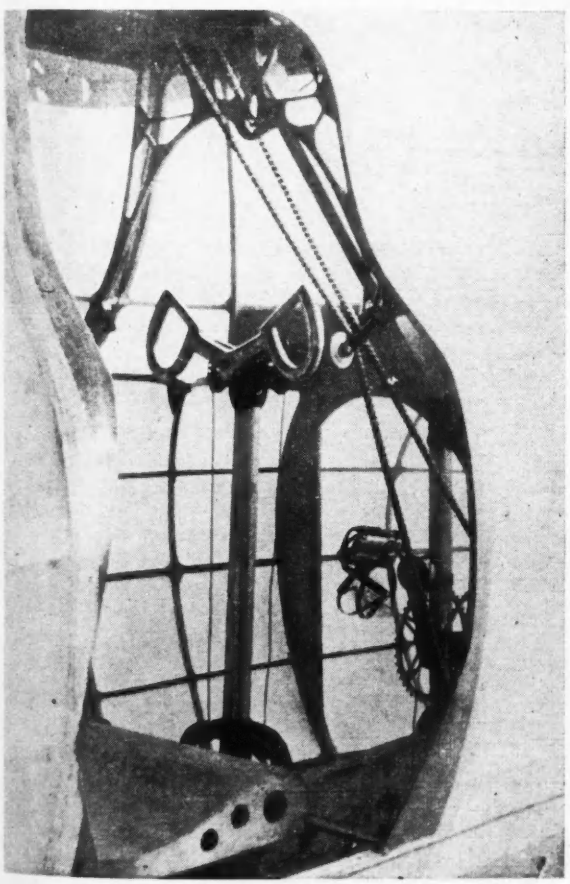

Figure 13
Cockpit of Bossi-Bonomi

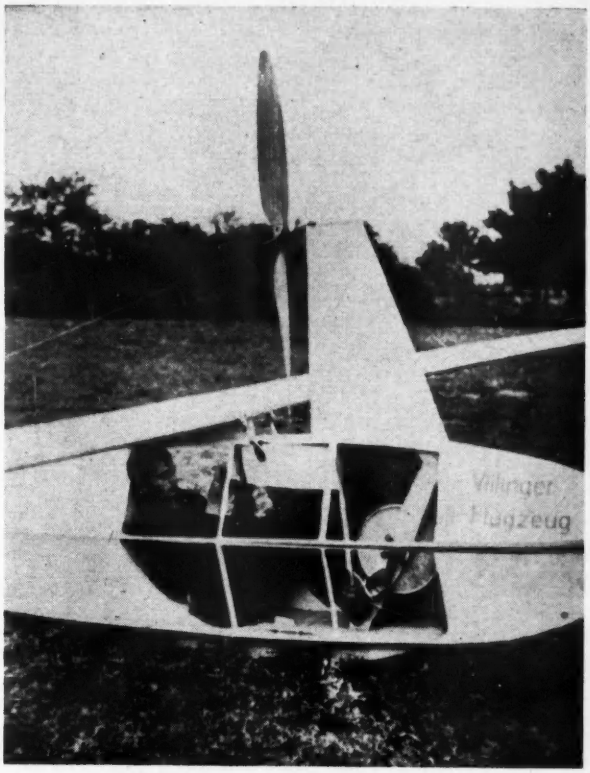

Figure 14
Cockpit of Haessler-Villinger.

Propeller drive

Although most inefficient, we have to start off with a pedal shaft which lies in the span-wise direction. The propeller inevitably has a shaft which is at right-angles to that and at a different level and at some distance from the initial drive shaft. One can use chains, belts, gears, connecting rods, and all of these have disadvantages. Chains and gears are heavy. Belts produce high bearing loads and therefore heavy mechanisms. Connecting rods, having shafts at right-angles, are mechanically impossible, although in practice they can be used by appropriate wangling.

Figure 13 shows the cockpit of the Bossi-Bonomi, the foot pedals being clearly seen. Figure 14 shows the cockpit of the Haessler-Villinger. The compact accommodation should be noted. In flight, only the head of the pilot is visible.

Flapping wings

The flapping wing can be considered as a propeller that does not go all the way round, and there is no reason to suppose that it is any more efficient than a properly designed propeller, except for its effective saving of the

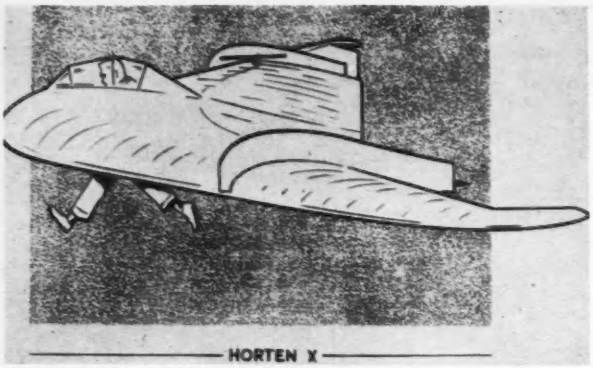

Figure 15
Sketch—Horten

profile drag of the propellers. It has, however, one virtue which is shared by no other means of propulsion, for, if properly designed, it need have no parasitic drag when not in use, and is therefore the obvious means of propelling an aircraft in which it is intended to soar part of the time, in other words, when used as an auxiliary source of power. For any aircraft which is to be power-driven at all times, I see no case for flapping wings. But I may not be seeing very clearly. Many talented people have done investigations on flapping wings and some full-scale flying experiments might certainly result in showing advantages which are at the moment not known to me.

But what of the man who wants an efficient sailplane he can manage all by himself? I refer to the modern "hang" type of glider exemplified by a recent Horten design, shown in Figure 15. Unlike Lilienthal and Pilcher and others, the modern hang glider is not controlled by swinging the body but by normal control surfaces. The problem is to get a structure that a man can carry (say 70 lb) and yet have a highly efficient aircraft into which the pilot's legs can be retracted while in flight. There is no doubt that this can be done and with spans of 40 ft or more, with or without tail. Here we have a different problem from that already discussed, for the maximum weight is fixed and the actual power required is not significant. The design problem is quite manageable, and there is no necessity for boundary layer control, although it might be considered as a refinement.

For such aircraft there may be a case for an auxiliary drive to extend the glide. Here is one application for the flapping wing if so designed that when not in use it has no extra drag. All or part of the wing may flap and it is a design problem to discover in a given case whether it is easier to flap a small part of the wing fast or more of it slowly.

Auxiliary manpower for sailplanes

The use of auxiliary manpower in a normal sailplane is not worth much because they need so much power. For instance, a Minimoa needs almost 3 hp to fly level and to fly level with full foot power (one minute limit) would require in addition an updraft of 1.6 fps.

CONCLUSION

In 1955 the development position regarding man-powered flight is like powered flight in 1900. A few people knew that flight was possible, but there were gaps in the knowledge, such as on wing section data, on light power plants, on control and stability.

Today we can see that if we stretch our knowledge and assurance very little, man-powered flight is possible. We need not be discouraged because it is apparently only possible at very low altitudes, at low speeds and at the expense of some complication and great physical effort. If we did not have to stretch ourselves to do this, it would long since have been done.

But once it has been done, it will be child's play, and the woods will be full of inventors turning out improvements and easements which will make this paper look silly.

As Major Baden-Powell said in 1902: "All that we require, then, to attain practical flight is to make a machine only slightly better than those already in existence. To accomplish this, all that is required is a certain amount of skill and a good deal of money."

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OPTIMUM STRUCTURAL DESIGN OF WING BOX BEAMS†

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SUMMARY

This paper analyses the following types of construction used for compression covers on wing box beams:—1) Skin and Z-section stringers, 2) Double skin and corrugation panels and 3) Solid plate multi-web construction. Curves of maximum compressive stress against the "Structural Index" are calculated and design data charts are derived in each case to help select the most efficient rib or web spacing as well as the actual dimensions of the panel elements.

In addition, the structural efficiency of the above types of construction is compared on the basis of:—1) Applied bending moment per inch width of structural chord, 2) Depth of the box beam and 3) Minimum skin thickness required for torsional rigidity.

Illustrative examples are shown in which the results of this investigation are applied to four different hypothetical airplanes.

INTRODUCTION

THE modern trend in wing structural design is towards the thick skin distributed flange box beam necessitated by the unbuckled surfaces and high torsional stiffness demanded for high speed.

For any given loading condition the material of a box beam is stressed in tension, compression and shear, and for an efficient design it must fulfil these functions in the most economical manner. The efficiency of the tension surface depends mainly on the quality of material available and this substantially applies to the shear webs, although efficiency here does depend somewhat on type of construction and loading. However it is in the compression surface that efficiency depends to the greatest extent on the type of construction, geometry of the elements and intensity of loading. We therefore find that in most cases the internal structural arrangement of the box beam depends on the manner in which the compression cover is designed.

This paper analyses several types of construction commonly used for compression covers and compares their efficiencies on the basis of the parameters generally governing the design of wing box beams. These are 1) applied bending moment per inch of structural chord, 2) depth of the beam and 3) minimum skin thickness required for torsional stiffness. In addition, once the general type of construction has been selected on the above basis, design charts are derived for each case to help select the most efficient rib pitch or web spacing as well as arrive at the optimum proportions of the elements of the compression cover itself.

It should be emphasized that the ideal design thus arrived at will not always be the most practical one, but it can serve as the starting point from which a practical design may ultimately emerge. Also the theoretical optimum is a useful datum against which the final design may be compared with from the structural efficiency point of view.

LIST OF SYMBOLS

A_s	stringer area—sq in.
B	spanwise web spacing—in.
b	stringer spacing—in.
c	width of stringer flange—in.
D	depth of box beam section—in.
d	depth of corrugations—in.
E	Young's Modulus of elasticity
e	width of corrugation flats—in.
f_b	initial instability stress of panel—lb/sq in.
f_{max}	maximum compressive stress—lb/sq in.
h	height of stringer—in.
K	efficiency factor
k_1	initial instability stress coefficient
L	rib spacing—in.
M	applied bending moment per in. structural chord-lb in/in.
N	number of cells in multi-web box
P	panel end load, lb/in.
p	corrugation pitch, in.
t, t_c, t_s	skin, corrugation and stringer thickness respectively, in.
W	unit weight of compression cover plus ribs (or webs), lb/sq in.
w_r, w_w	unit weight of ribs and webs respectively, lb/sq in.
ρ	density of material, lb/in.^{3}
η	reduction coefficient for E above proportional limit
θ	angle of trapezoidal corrugations, degrees

TYPES OF COMPRESSION COVER CONSTRUCTION

The various types of construction generally found in wing box beams can be classified into two broad

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categories depending on the manner in which the compression cover is supported against instability. These are 1) Wide column construction and 2) Plate construction.

The wide column construction consists of skin and stringer panels together with chordwise ribs at regular intervals. Each stringer with its effective skin is considered as an individual column, supported at the ribs but receiving negligible support from the adjacent stringers or shear webs because of the small chordwise stiffness of the covers. Several shapes of the stiffener are possible and used in practice, these may be Z, Y or hat section but the aim of them all is to get the maximum effective depth while at the same time avoiding local or torsional instability. The Z-stringer, while it may show a slightly lower efficiency than some of the other shapes, appears to be the most popular with designers as it has many practical advantages in manufacturing.

The plate type of construction consists of either solid plate or double skin sandwich covers supported only by spanwise webs. Because of the finite chordwise stiffness of the cover in this case, it acts as a plate of width equal to the web spacing. Fundamentally a plate is more efficient than a column but in practice, unless the loading is extremely high, the available material must be "expanded" into some form of double skin panel to provide overall stability, and the allowable compression stress may thus be limited by local buckling of the elements. The double skin panels used for these applications may have either high strength cores e.g. corrugations which carry part of the axial load as well as providing shear rigidity and buckling support for the skins, or may use cores of low density material e.g. honeycomb or end grain balsa to provide only shear rigidity and buckling support for the facings.

Double skin panels because of the greater manufacturing difficulties are not as extensively used as skin stringer panels, although corrugated sandwich covers have been successfully employed by a number of manufacturers. With the improvement of bonding techniques it is to be expected that honeycomb sandwich construction will increase in popularity, especially in view of the potential saving in manufacturing costs possible with this type of construction.

This paper will limit itself to a comparison between skin and Z-section stringer panels as representative of the column type of construction and double skin corrugation panels as well as solid plate panels as representative of the plate type of construction. It will be assumed that the skin remains unbuckled up to ultimate load and that the covers take the entire end load from bending.

MAXIMUM COMPRESSIVE STRESS

It has been shown in References 1 and 2 that the maximum compressive stress that can be developed in a given panel may be expressed theoretically by the following relationship:

$$f_{max.} = K\sqrt{\frac{P}{L}\eta E} \qquad (1)$$

where the term P/L (or P/B in the case of plate construction) is the fundamental loading parameter generally referred to as the "Structural Index", and K is an

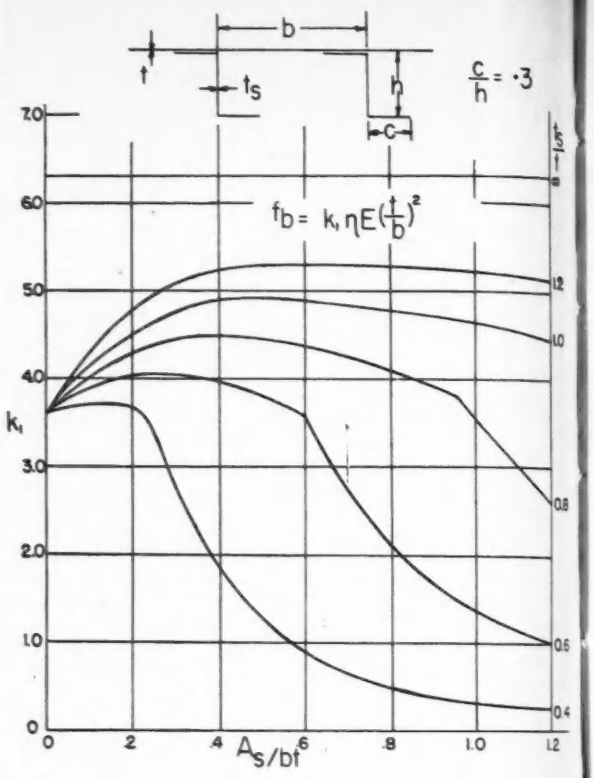

Figure 1
Initial instability stress for panels with Z-section stiffeners.

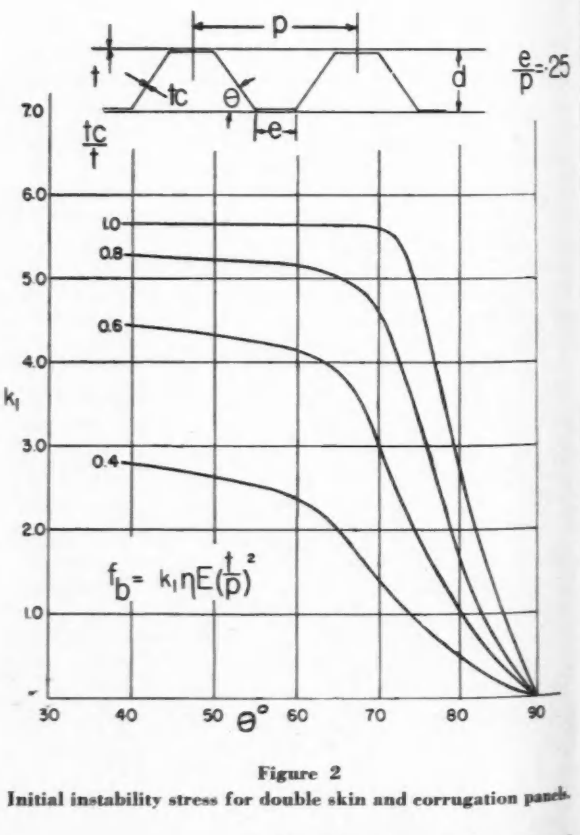

Figure 2
Initial instability stress for double skin and corrugation panels.

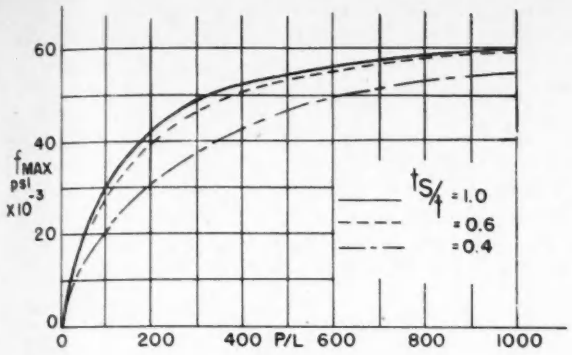

Figure 3

Maximum compressive stress vs P/L for panels with Z-section stiffeners (75S-T6 Aluminum alloy)

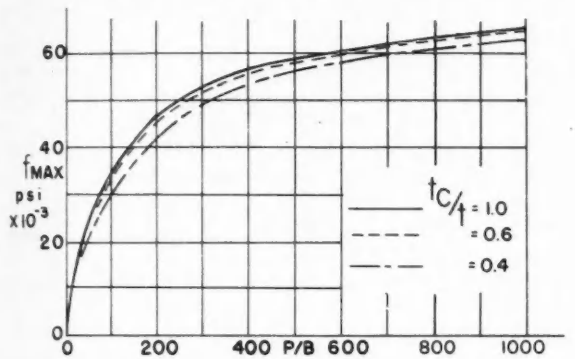

Figure 4

Maximum compressive stress vs P/B for double skin and corrugation panels (75S-T6 Aluminum alloy)

"Efficiency Factor" which depends only on the geometrical proportions of the panel.

For any given panel the equation for maximum stress can be derived by equating the average stress on the section to the column stress and local instability stress. This is based on the assumption that the maximum stress, or optimum design is obtained when the various modes of failure coincide. The difficulty, from the theoretical point of view, is to express the local instability stress in terms of the geometrical parameters of the panel, as the solution of this problem must consider not only the various instability modes possible for the given panel but also the interaction between them.

This problem has been solved in Reference 3 for the case of flat panels with Z-section stringers by considering the torsional and local instability modes of the stringer in combination with buckling of the skin, and typical results are given in Figure 1. A solution has also been found in Reference 4 for double skin and corrugation panels in which the critical combinations of local instability modes of the corrugation and skin buckling are considered. Typical results are shown in Figure 2.

With the above data available, the "Efficiency Factor" K is readily calculated in terms of the geometrical parameters of the panel, and the maximum compressive stress as given by Eq. (1) may be plotted against the "Structural Index". The actual derivations are given in Appendix I and the results for the types of construction considered here, in 75S-T6 Material, are shown in Figures 3, 4 and 5. The method presented here is however quite general and can be applied to any type of construction or material.

It is to be noted that in the case of the skin-stringer and double skin corrugation panels, the results are given for several ratios of stringer (or corrugation) to skin thickness. This will be made use of at a later stage to evaluate the reduced efficiency when thicker than optimum skin gauges are required for torsional stiffness of the box beam.

The maximum compressive stress calculated in this section is a theoretical value which may not always be reached in actual practice. The local instability stresses are reduced by initial waviness of the plate or stiffeners, and a bow in the stringer or other eccentricities may cause flexural collapse to occur prior to the calculated column load. Also a continuous line of contact, fixed in space, has been assumed between the skin and stringer. This implies relatively strong rivet attachments or some form of bonding in addition to sturdy stringers.

The latter condition becomes very important with thick plate construction where the ratio of stringer to skin thickness may be as low as 0.4 and a large reduction in the skin buckling stress may occur due to the relatively thin stringer flange behaving as an elastic foundation. These factors must be considered in the detail design and tests should be carried out for panels that fall in this range.

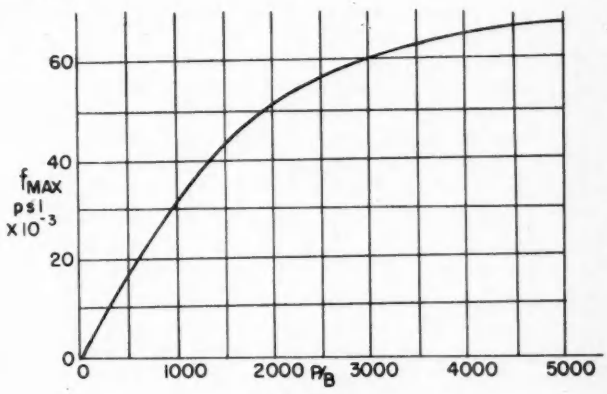

Figure 5

Maximum compressive stress vs P/B for solid plate panels (75S-T6 Aluminum alloy)

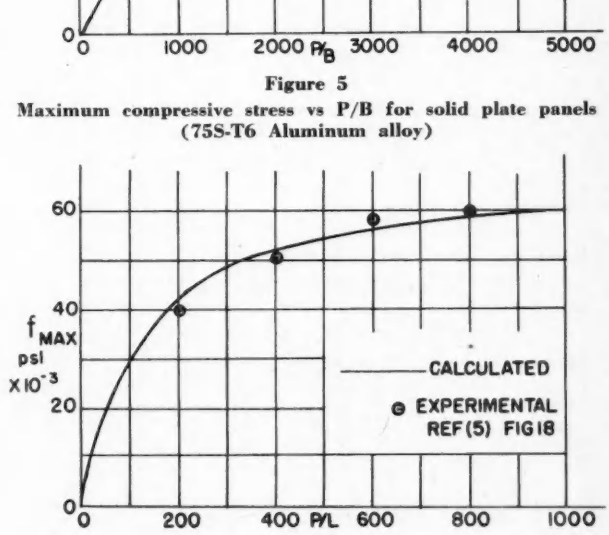

Figure 6

Comparison of experimental and calculated results for panels with Z-section stiffeners (75S-T6 Aluminum alloy)

With the double skin corrugation panels this last effect is probably not as important as the crests of the corrugation are more rigid than the free flanges of the stringers.

A comprehensive test program, covering a very wide range of variables, has been carried out by NACA[5] on flat compression panels stiffened by extruded Z-section stringers. The results from the above tests are compared in Figure 6 with the analytical results calculated here and show quite good agreement.

Unfortunately no such complete test data is available for double skin and corrugation panels, with which the theoretical results derived here can be compared.

WEIGHT OF IDEALIZED BOX BEAM

Having calculated curves of maximum allowable stress vs "Structural Index" for the various types of construction considered, it is now possible to obtain a weight comparison of these constructions when used in box beams.

In this section the variation of weight for a box beam with respect to the loading and the geometry is derived under the following assumptions:

1) The box section is rectangular.
2) The compression cover is simply supported by either uniformly spaced chordwise ribs or spanwise webs and is subjected to a uniform axial load intensity.
3) The effect of transverse air load and shear stress due to torsion on the strength is neglected.
4) The tension cover is designed to a constant stress irrespective of the loading, geometry or type of construction. Thus its weight has no effect on optimum considerations or comparative results and may be neglected.
5) All the direct shear is taken by the two end webs of the beam. These, as above, are designed to a constant stress and thus their weight is neglected.

The weight used for the purpose of this analysis therefore consists only of the compression cover and internal supporting structure of the box.

For the stringer and rib construction, the weight of the compression cover plus ribs, per unit area of surface is,

$$W = \frac{\rho P}{f} + w_r \frac{D}{L} \tag{2}$$

As can be seen this expression is independent of the structural chord, the first term representing the cover weight and the second term the distributed rib weight per unit surface area.

In the case of the plate type of construction the combined cover and web weight per unit surface area is,

$$W = \frac{\rho P}{f} + w_w \frac{D}{B}\left(1 - \frac{1}{N}\right) \tag{3}$$

The above expression is similar to Eq. (2) except for the additional variable N which makes it dependent on the chord. Actually the term $(1 - 1/N)$, which represents a reduction in internal web weight, appears because in this case the end webs contribute to supporting the cover as well as carrying the shear. In the limit when $N = 1$, i.e. no intermediate webs, the weight of supporting structure is zero and as N increases the term $(1 - 1/N)$ approaches unity indicating no reduction in weight of the supporting structure.

In actual practice the number of cells (i.e. N) will vary from 2 or 3 for most sandwich type covers to about 4 for solid plate covers, therefore for the purpose of this analysis it may be assumed that $N = 3$ in the former case and $N = 4$ in the latter. Any deviation from these values will not introduce serious errors in the results and with this assumption Eq. (3) becomes independent of the structural chord and the weight of the box beam may be compared directly with that given by Eq. (2) for the stringer construction.

Before the optimum weight calculations can be completed it is necessary to determine the rib and web weights. Due to the complex nature of the loads on a rib or web, the variation in detail design and minimum size considerations by which most ribs are designed, their weight cannot be predicted on purely theoretical

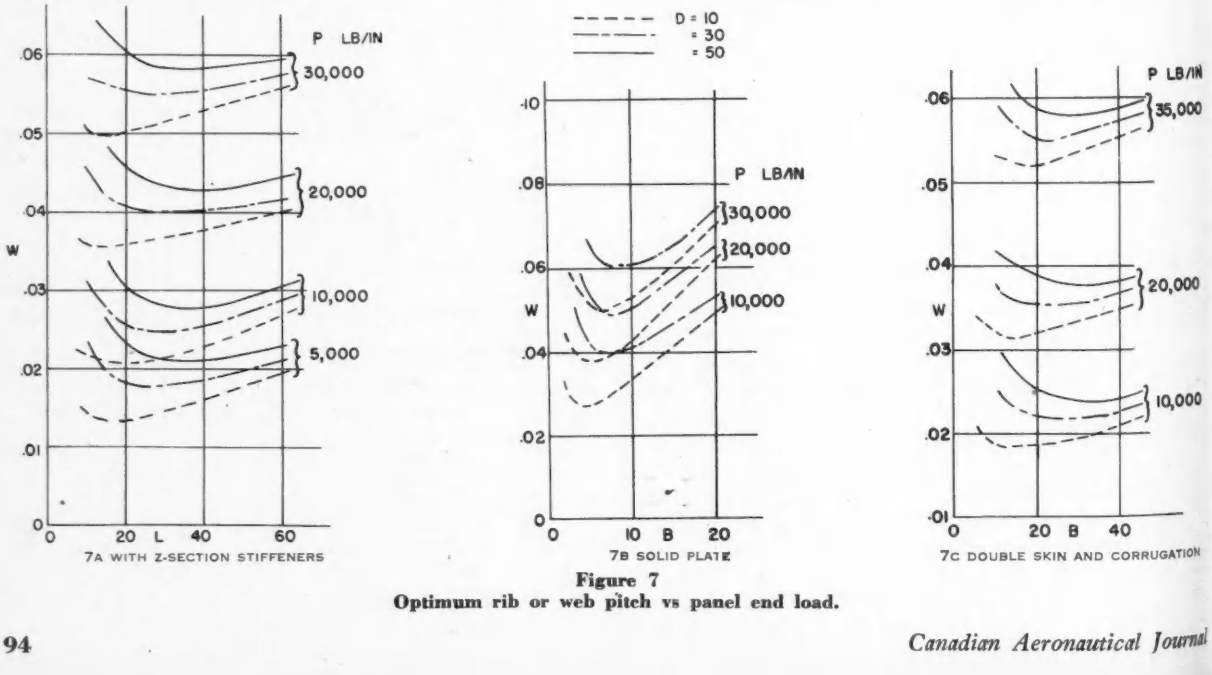

Figure 7
Optimum rib or web pitch vs panel end load.

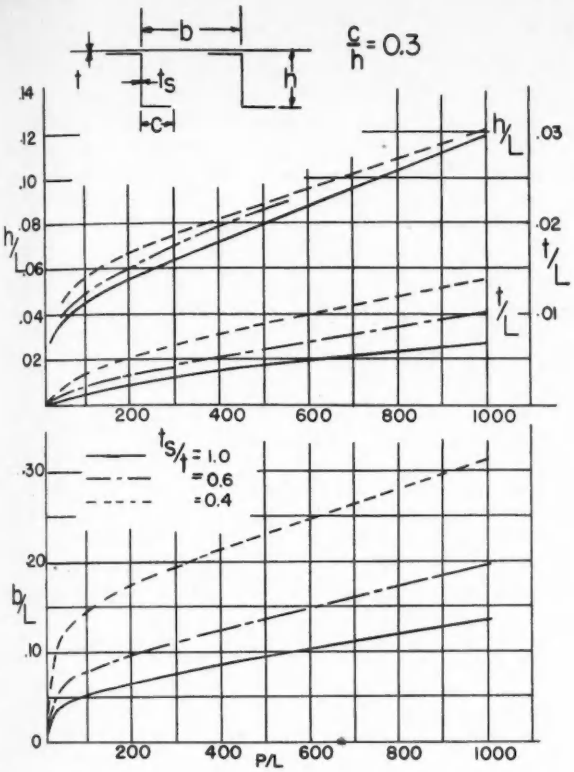

Figure 8

Design charts for optimum panels with Z-section stiffeners.

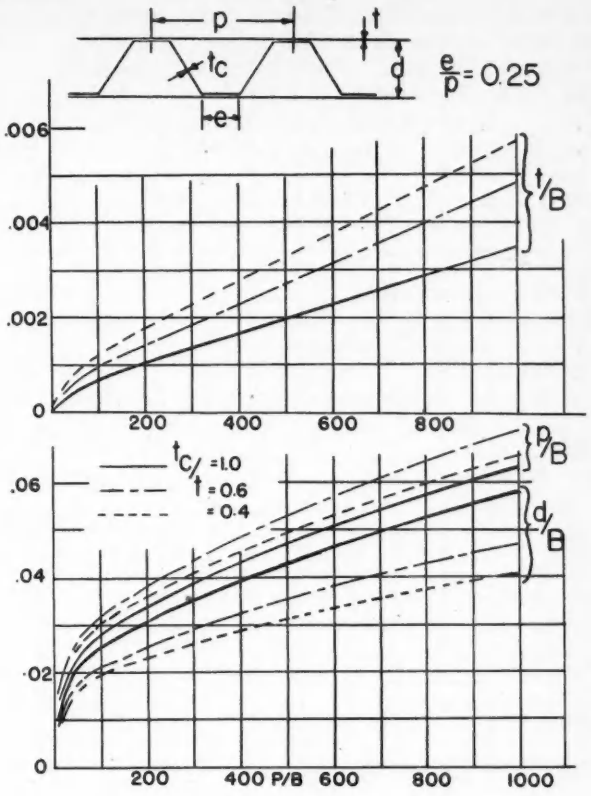

Figure 9

Design chart for optimum double skin and corrugation panels.

grounds. Investigation into actual rib weights of several airplanes from different manufacturers indicate that the weight per unit area of rib surface is approximately constant irrespective of the loading or size, and that this figure is approximately .005 lb/sq in.

Very little actual data is available for the weight of spanwise webs but, assuming them to be made up of some form of double skin sandwich construction, a brief investigation indicates that the same figure as above i.e. .005 lb/sq in will meet practical minimum size and buckling considerations as well as the maximum loads likely to be imposed in the range considered here.

Therefore for the purpose of this analysis a constant value of .005 lb/sq in will be used for both rib and web weights.

OPTIMUM RIB AND WEB SPACING

It has been previously shown that the maximum compressive stress is proportional to the Structural Index, P/L or P/B. Therefore, for a given value of P a reduction in the rib or web spacing (L or B) increases the allowable stress, thereby saving weight in the covers. However reducing the spacing increases the weight of ribs or webs and thus an optimum pitch must be found for any given loading and depth of section at which the combined weight of cover and supporting structure is a minimum. This can be done by differentiating Eqs. (2) and (3) with respect to L and B respectively. In the elastic range f may be replaced by $\sqrt{PE/L}$ and the expressions easily differentiated analytically. However in the inelastic range where E is no longer a constant, the differentiation must be done by graphical means i.e. plotting the weight against rib pitch for various values of P and D.

The variation of total weight against rib or web pitch for various values of P and D, thus obtained is shown plotted in Figure 7. It can be seen that an optimum rib or web spacing exists for any given design condition although the weight penalty involved in departing from the optimum is not very large, specially if the pitch is made larger. This is fortunate because in any actual design it is always desirable to keep the rib or web spacing as large as possible for manufacturing and fuel stowage reasons. The maximum stress is also thereby reduced, which is beneficial from rigidity and fatigue considerations.

OPTIMUM PANEL PROPORTIONS

The optimum proportions of a panel can be readily expressed in terms of the Structural Index and Efficiency Factor for the type of design used. The derivations are given in Appendix II and Figures 8 and 9 show the resulting stringer and corrugation panel proportions plotted against the Structural Index for various ratios of stringer (or corrugation) to skin thickness.

Starting with a given end load in the panel, the optimum rib or web spacing can be determined from Figure 7, and hence the Structural Index. The actual panel dimensions are then readily obtained from the above design charts. It is important to remember that these dimensions may have to be modified to take into account manufacturing or detail design considerations.

The effect of such modifications on the overall efficiency however is generally not very great.

Up to this stage we have seen how to determine the most efficient panel design for each type of construction and a given loading by finding the optimum rib spacing, stringer spacing, corrugation pitch, depth etc. The following section will complete the "optimizing process" by comparing the relative efficiency of each type of construction on a common basis and thus help select the most efficient structure for a given set of conditions.

WEIGHT COMPARISON CURVES

By substituting the optimum rib or web spacing from Figure 7 into Eqs. (2) and (3), the optimum weight of a section can be calculated as a function of the endload in the panel and the depth of the section. Figure 10 shows typical results for the types of construction considered here.

To be of greatest value to the designer, however, it is necessary to carry out the comparison on the basis of bending moment on the section and equivalent skin

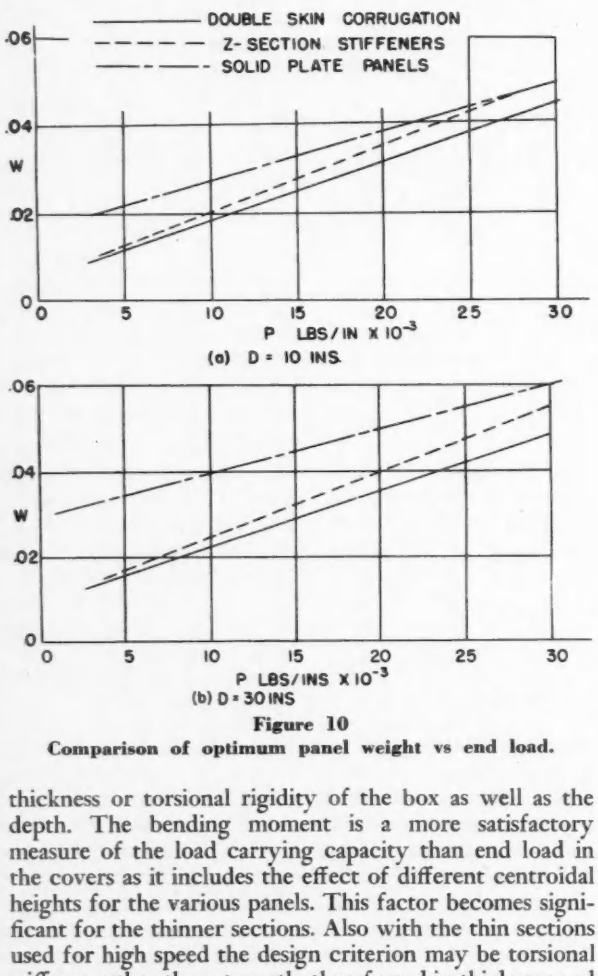

Figure 10
Comparison of optimum panel weight vs end load.

thickness or torsional rigidity of the box as well as the depth. The bending moment is a more satisfactory measure of the load carrying capacity than end load in the covers as it includes the effect of different centroidal heights for the various panels. This factor becomes significant for the thinner sections. Also with the thin sections used for high speed the design criterion may be torsional stiffness rather than strength, therefore skin thickness and effective torsion box area must be included in the comparisons.

The end load in the panel is converted into bending moment readily enough by multiplying by the centroidal depth as determined from the panel geometry. In order to introduce the skin thickness as a variable, it is necessary to repeat the whole process of finding the optimum weight for each of the several ratios of stiffener to skin thickness by using the appropriate maximum stress curves from Figures 3 and 4. When the results for each ratio of t_s/t are plotted together it is possible to draw in lines of constant skin thickness, as it can be shown that each point on the resulting curves corresponds to a fixed value of skin thickness.

Figure 11 summarizes the above calculations and shows a comparison of the various constructions on the basis of 1) bending moment per inch chord, 2) depth of section and 3) equivalent skin thickness. The equivalent skin thickness in the case of the double skin and corrugations takes into account the shear rigidity of the corrugations as well as the reduction of effective torsion box area.

The charts of Figure 11 show the double skin construction to be the most efficient over the entire range of variables on a purely strength basis, although the skin and stringer panels become almost as efficient at the lower loadings. The solid plate construction is the most efficient in the thinner sections where high torsional rigidity is required.

ILLUSTRATIVE EXAMPLES

Table 1 shows a comparison of the relative weights of compression cover plus supporting structure for each of the types of construction investigated here, in the case of four different hypothetical airplanes.

In the case of Airplane A, which is a small feeder line transport, the double skin and corrugation cover is about 17% lighter than the skin and stringers at the root section.

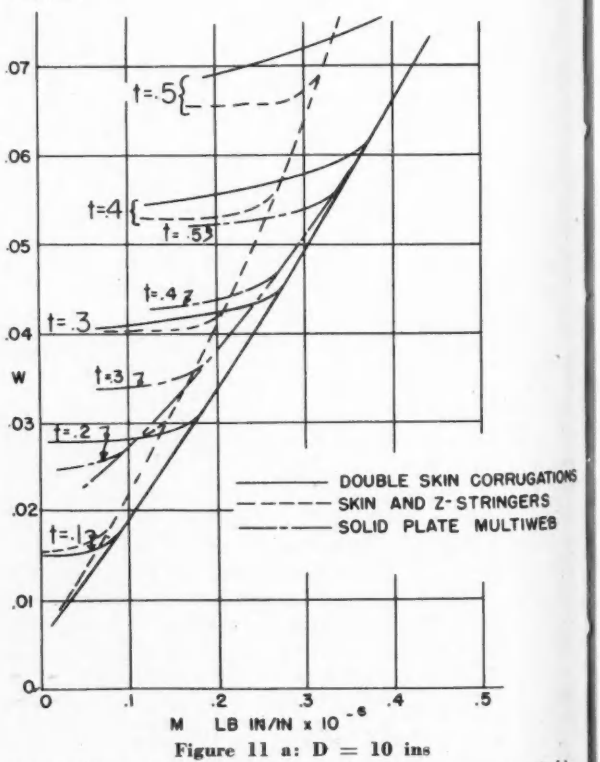

Figure 11 a: D = 10 ins
Comparison of optimum weight vs bending moment and skin thickness.

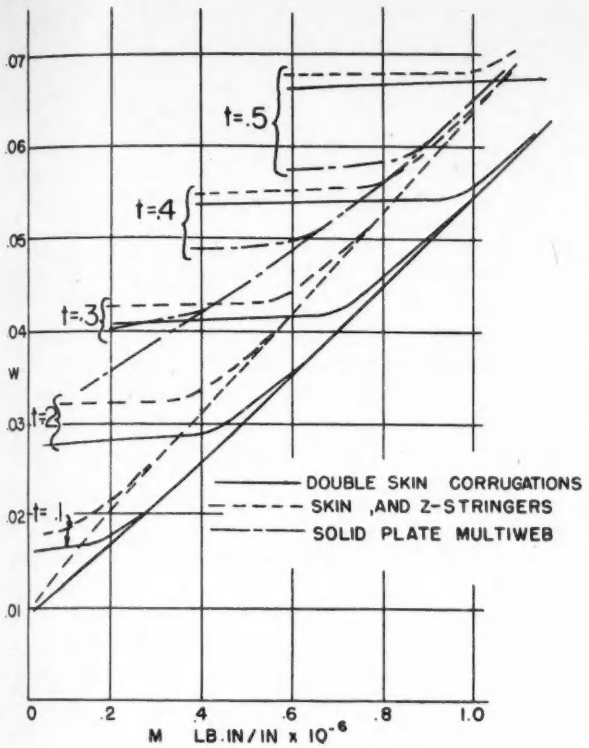

Figure 11 b: D = 30 ins

Comparison of optimum weight vs bending moment and skin thickness.

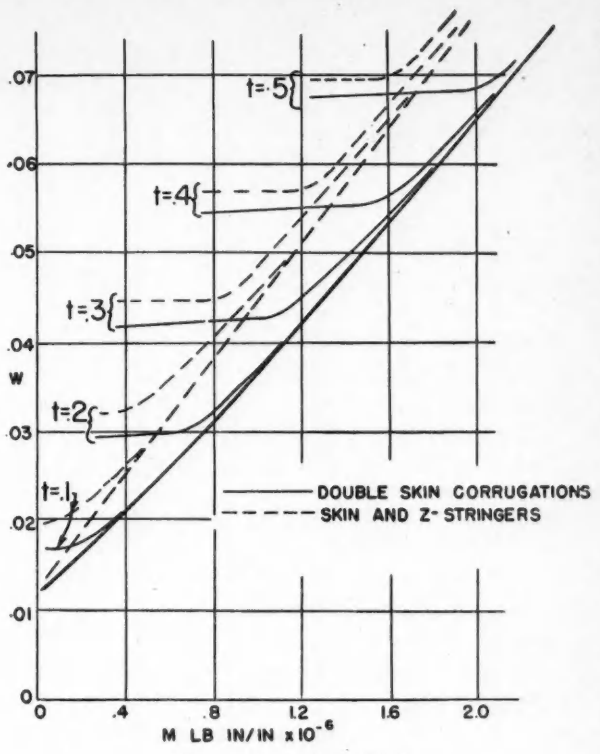

Figure 11 c: D = 50 ins

Comparison of optimum weight vs bending moment and skin thickness.

However a detailed design study shows, in this particular case, that the saving on the total wing weight amounts to only 1.5% or 60 lb. A consideration of the manufacturing costs involved would probably lead to a choice of the skin and stringer construction in this case.

Airplane B is a large turbo-prop transport with a cruising speed of about Mach. 0.6. Here the double skin construction shows a weight saving of 23% over the skin and stringers at the root section. The saving on total wing weight in this case amounts to about 4% or 500 lb, which makes the double skin and corrugation construction seem worthwhile.

Airplane C is a large jet transport cruising at about Mach. 0.75. In this case the double skin construction is again the lightest, but as the covers here are designed by the stiffness criterion, the relative weight advantage is less because of the reduction in effective torsion box area on the thin wing section with the double skin panels. Also in this case the fuel stowage problem is very critical and practically dictates the use of integral fuel tanks with skin and stringer construction in order to obtain the largest usable volume by having the fuel extend to the skin line.

In the case of Airplane D the solid plate multi-web construction is 18% lighter than the skin and stringers and 25% lighter than the corrugation covers. Here the multi-web construction can probably be used to good advantage.

CONCLUSIONS

The optimum panel design for several types of construction used in wing box beams has been determined,

TABLE 1

COMPARISON OF WING WEIGHT (COMPRESSION COVER AND SUPPORTING STRUCTURE) FOR FOUR DIFFERENT AIRPLANE CATEGORIES.

1) Airplane Category	A Small Transport	B Large Transport	C Jet Transport	D Supersonic Fighter
2) Wing Loading (lb/sq. ft.)	35	75	80	90
3) Wing Root Thickness	20% (30 in.)	16% (40 in.)	10% (30 in.)	7% (10 in.)
4 Bend. Moment/in Chord (lb in/in)	180,000	700,000	850,000	300,000
5) Required Skin Thickness (in)	0.08	0.20	0.40	0.50
6) Weights—Skin-Stringers	.018	.044	.060	.066
Corrugations	.015	.034	.055	.072
Solid Plate	–	–	.060	.054

and the relative efficiency for these constructions compared on a common basis.

Results are shown for skin and Z-section stringer panels, solid plate panels and double skin corrugation panels, and indicate the latter to be the most efficient except for very thin wings or low loading. For thin wing sections where the covers are designed by stiffness requirements the solid plate multi-cell construction is the most efficient, while at very low loading the skin and stringer panels are practically equal to the corrugation covers. The above results are obtained by assuming use of 75S-T6 aluminum alloy throughout, but the method is quite general and can be applied to any material as well as panel design.

Illustrative examples are presented and show how practical considerations may influence the choice of construction.

APPENDIX I.

DERIVATION OF EQUATIONS FOR MAXIMUM COMPRESSIVE STRESS

1) Skin and Z-Stringer Panels

The initial instability stress is given by:

$$f_b = k_1 \eta E \left(\frac{t}{b}\right)^2 \tag{1}$$

where k_1 is obtained from Figure 1.

The critical column stress is given by:

$$f_{cr} = \pi^2 \eta E \left(\frac{\rho}{L}\right)^2 \tag{2}$$

where ρ is the radius of gyration of the section.

For the purpose of this analysis η will be assumed equal to $\sqrt{\tau}$ and τ in Eqs. (1) and (2) respectively, where τ is the ratio of Tangent Modulus to Elastic Modulus. See Reference 6.

The average applied stress in the panel is:

$$f_a = \frac{P}{\bar{t}} \tag{3}$$

where $\bar{t}$ is the equivalent thickness of the total panel material.

Expressing ρ and $\bar{t}$ in terms of a panel dimension and a non-dimensional coefficient.

$$\rho = k_2 b \tag{4}$$

$$\bar{t} = k_3 t \tag{5}$$

where

$$k_2 = \left[\frac{t}{t_s}\right]\left[\frac{A_s}{bt}\left(\frac{1}{1+2\frac{d}{h}}\right)\right]\left[\frac{\frac{1}{4} - \frac{1}{4\left(\frac{A_s}{bt}+1\right)} + \frac{A_s}{bt}\left(\frac{\frac{1}{12}+\frac{1}{2}\frac{d}{h}}{1+2\frac{d}{h}}\right)}{\frac{A_s}{bt}+1}\right]^{\frac{1}{2}} \tag{6}$$

and

$$k_3 = \frac{A_s}{bt} + 1 \tag{7}$$

For an optimum panel design,

$$f_{max.} = f_a = f_b = f_{cr} \tag{8}$$

Therefore substituting Eqs. (4) and (5) into Eqs. (2) and (3) and eliminating the variables t and b by multiplying Eq. (1) by Eq. (2) × Eq. (3)^2, we obtain the expression,

$$f_{max.} = K\sqrt{\frac{P}{L}\tau^{\frac{3}{4}}E} \tag{9}$$

where

$$K = \sqrt{\pi}\sqrt[4]{\frac{k_1 . k_2^2}{k_3^2}} \tag{10}$$

The optimum value of the Efficiency Factor, K for a given ratio of t_s/t is obtained from Eq. (10) by plotting K against A_s/bt (assuming $d/h = 0.3$).

The optimum values of K are 0.95, 0.87 and 0.69 for $t_s/t = 1.0$, 0.6 and 0.4 respectively. Eq. (9) is shown plotted in Figure 3 for 75S-T6 aluminum alloy using the above values for K.

2) Double Skin and Corrugation Panels

The initial instability stress is given by:

$$f_b = k_1 \eta E \left(\frac{t}{p}\right)^2 \tag{11}$$

where k_1 is obtained from Figure 2 and η is assumed equal to $\sqrt{\tau}$ as previously.

The critical instability stress for a non-isotropic plate is given in Reference 7 as:

$$f_{cr} = \frac{2\pi^2}{B^2 t}\left(\sqrt{D_X . D_Y} + D_3\right) \tag{12}$$

where D_X, D_Y and D_3 are the flexural and torsional rigidities of the plate as defined in Reference 8.

The average applied stress on the section is:

$$f_a = \frac{P}{\bar{t}} \tag{13}$$

The last term of Eq. (12) may be expressed in the following form:

$$\sqrt{D_X . D_Y} + D_3 = k_2 \tau^{\frac{1}{2}} E \bar{t} p^2 \tag{14}$$

where

$$k_2 = \frac{\left(1-2\frac{e}{p}\right)^2 \tan.^2\theta\left[.565 + \sqrt{.303 + .550\frac{t_c}{t}\left(\frac{e}{2p} + \frac{1-2\frac{e}{p}}{12\cos.\theta}\right)}\right]}{4\left[2 + \frac{t_c}{t}\left(\frac{1-2\frac{e}{p}}{\cos.\theta} + 2\frac{e}{p}\right)\right]} \tag{15}$$

and

$$\bar{t} = k_3 t \tag{16}$$

where

$$k_3 = 2 + \frac{t_c}{t}\left(\frac{1-2\frac{e}{p}}{\cos.\theta} + 2\frac{e}{p}\right) \tag{17}$$

Proceeding now in a similar manner as for the skin and stringer panels,

$$f_{max.} = K\sqrt{\frac{P}{B}\tau^{\frac{1}{2}}E} \tag{18}$$

where

$$K = \sqrt{\pi}\sqrt[4]{\frac{2k_1 . k_2}{k_3^2}} \tag{19}$$

The optimum value of the Efficiency Factor K, for the double skin and corrugation panels is calculated in the same manner as in the previous section (assuming $e/p = 0.25$) and is equal to 1.07, 1.05 and .95 for value of $tc/t = 1.0$, 0.6 and 0.4 respectively.

Eq. (18) is plotted in Figure 4 for 75S-T6 aluminum alloy using the above values.

3) Solid Plate Panels

The initial instability stress is given by:

$$f_b = k_1 \tau^{\frac{1}{2}} E \left(\frac{t}{B}\right)^2 \tag{20}$$

also

$$f_a = . \frac{P}{t} \tag{21}$$

Mulitply Eq. (20) by Eq. (21)^2 and put $f_b = f_a = f_{max}$

$$f_{max.} = K\sqrt[3]{\left(\frac{P}{B}\right)^2 \tau^{\frac{1}{2}} E} \tag{22}$$

where

$$K = \sqrt[3]{k_1} = \sqrt[3]{3.62}$$

Eq. (22) is plotted in Figure 5 for 75S-T6.

APPENDIX II.
DERIVATION OF OPTIMUM PANEL PROPORTIONS

1) Skin and Z-Stringer Panels

From Eq. (3) and (5)

$$f = \frac{P}{k_3 t} \tag{23}$$

Substituting above expression for f in Eq. (9) and multiplying both sides by L,

$$\frac{t}{L} = \frac{1}{K k_3}\sqrt{\frac{P}{L\tau^{\frac{3}{4}}E}} \tag{24}$$

From Eqs. (1) and (9),

$$k_1 \tau^{\frac{1}{2}} E\left(\frac{t}{b}\right)^2 = K\sqrt{\frac{P\tau^{\frac{3}{4}}E}{L}}$$

$$\frac{t}{b} = \sqrt{\frac{K}{k_1}}\sqrt[4]{\frac{P}{L\tau^{\frac{3}{4}}E}} \tag{25}$$

Dividing Eq. (24) by Eq. (25)

$$\frac{b}{L} = \sqrt{\frac{k_1}{K^3 k_3^2}}\sqrt[4]{\frac{P}{L\tau^{\frac{5}{4}}E}} \tag{26}$$

From Figure 1, the stringer area

$$A_s = ht_s\left(1 + 2\frac{d}{h}\right) \tag{27}$$

$$\therefore \frac{h}{b} = \frac{\frac{A_s}{bt}}{\frac{t_s}{t}\left(1 + 2\frac{d}{h}\right)} \tag{28}$$

Multiplying Eq. (26) by Eq. (28)

$$\frac{h}{L} = \frac{\frac{A_s}{bt}}{\frac{t_s}{t}\left(1 + 2\frac{d}{h}\right)}\sqrt{\frac{k_1}{K^3 k_3^2}}\sqrt[4]{\frac{P}{L\tau^{\frac{5}{4}}E}} \tag{29}$$

By substituting the optimum values of K and corresponding A_s/bt for each value of t_s/t in Eqs. (24), (26) and (29), the optimum panel proportions can be plotted against P/L as shown in Figure 8.

2) Double Skin and Corrugation Panels

The derivation follows in a similar manner as for the skin and stringer panels.

From Eq. (13) and (16)

$$f = \frac{P}{t} = \frac{P}{k_3 t} \tag{30}$$

Substituting Eq. (30) in Eq. (18) and multiplying by B

$$\frac{t}{B} = \frac{1}{K k_3}\sqrt{\frac{P}{B\tau^{\frac{1}{2}}E}} \tag{31}$$

From Eqs. (11) and (18)

$$\frac{t}{p} = \sqrt{\frac{K}{k_1}}\sqrt[4]{\frac{P}{B\tau^{\frac{1}{2}}E}} \tag{32}$$

Dividing Eq. (31) by Eq. (32)

$$\frac{p}{B} = \sqrt{\frac{k_1}{K^3 k_3^2}}\sqrt[4]{\frac{P}{B\tau^{\frac{1}{2}}E}} \tag{33}$$

From Figure 2, $\frac{d}{p} = \left(\frac{1}{2} - \frac{e}{p}\right)\tan.\theta$ (34)

Multiplying Eq. (33) by Eq. (34)

$$\frac{d}{B} = \left(\frac{1}{2} - \frac{e}{p}\right)\tan.\theta\sqrt{\frac{k_1}{K^3 k_3^2}}\sqrt[4]{\frac{P}{B\tau^{\frac{1}{2}}E}} \tag{35}$$

The optimum panel proportions are plotted in Figure 9 from Eqs. (32), (33) and (34) in a similar manner as for the skin and stringer panels.

APPENDIX III

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A PROGRESS REPORT ON THE CANADIAN AERONAUTICAL ENGINEER†

by R. B. Ferris* and S/L G. B. Waterman**

CURIOSITY is a fine thing if kept under control, but it can lead to a lot of hard work.

The chance remark, "Whatever happened to old so and so", started the whole thing off. It gave us the idea that it would be interesting to find out just what our college friends and confreres were doing and where they were doing it. Then we started wondering about their financial welfare and how they felt about the aviation industry. The next logical step was a survey to gather this information and from there we were drawn inexorably into writing a paper which would present a picture of at least one section of Canada's aviation industry.

THE APPROACH

The University of Toronto (which gives the only Canadian degree in aeronautical engineering) was asked to supply addresses for the graduating classes of 1948-49-50 in aeronautical engineering. These classes were chosen since the number of graduates reached a peak in this period and since most of them were friends or acquaintances of the writers and would, therefore, have a personal interest in the survey. In the meantime, a questionnaire was prepared with the assistance and helpful criticism of Mr. H. C. Luttman, Secretary of the C.A.I., and Mr. J. P. Francis, of the Department of Labour. Mr. Etkin's paper to the C.A.I., in Toronto last year was used as a guide in listing the many fields of employment for aeronautical engineers. In retrospect it must be admitted, however, that the questionnaire still contained several weaknesses which detracted somewhat from its effectiveness.

We would also like to acknowledge here the assistance by Mr. Redmayne of the Air Industries and Transportation Association in providing secretarial and mailing facilities for the original questionnaire and for the follow-up letter which was sent out to the tardy or recalcitrant candidates. Our thanks are also due, to the illustrating section of the Air Materiel Command headquarters for its excellent reproduction of the charts and tables.

THE RESPONSE

There were 129 graduates in aeronautical engineering between 1948 and 1950, and of these 127 are still living. Replies were received from 93 of these i.e., 73% of the total. By year, these replies were as follows:

1948	–	16	(73%)
1949	–	41	(76%)
1950	–	36	(71%)

Questionnaires were returned from as far away as Hucclecote, England; Trondheim, Norway; and Los Angeles, California; while some of those who were living in Ottawa never did get around to sending in their replies. The 93 engineers who replied can be divided into the following groups:

Canadian aircraft industry	43
American aircraft industry	10
Canadian military services	22
Employed outside North America	6
Not employed in aviation	12

Since this was to be a paper about people in the aircraft industry, the twelve graduates who had left aviation were not considered in analysing the returns other than when determining such vital statistics as age, marital status, etc.

In considering the economic situation an additional four engineers working outside North America were dropped because their salary scales could not readily be compared with those on this continent. This means that economic statistics have been based on a total of 77 of the 93 replies received. To present an accurate and realistic picture of the young aeronautical engineer in Canada, it would have been preferable to have had more replies for study; however, it was felt that the statistics obtained, while sometimes incomplete, were fairly representative of the graduate's present position.

FACTS AND FIGURES

Having extracted all the figures from the questionnaire we found that we had created a hypothetical person – the average aeronautical engineer. This mathematical monstrosity is now 31.7 years old, married, and has 1.56 children. He is a veteran with over three years active service in the Second World War, does not have post-graduate training and is a member of the C.A.I.

Returning to reality, a study of the graduating ages over the three year period proved rather interesting. Figure 1 is a histogram of graduating ages which also shows the mean graduating age for each of the three years. Starting with the class of 1948 these means were 24.3, 25.8 and 26.7 years respectively – a very definite increase each year. This and the positive skewness of the histogram are undoubtedly due to the influence of

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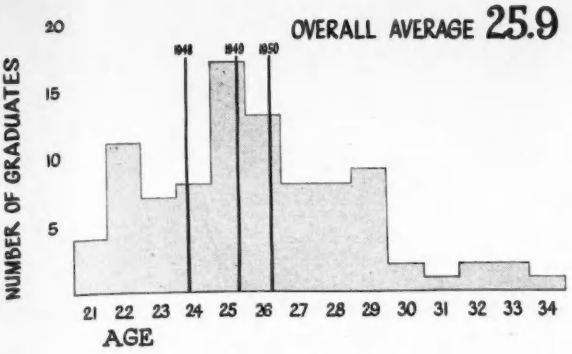


Figure 1
Age on Graduation

veterans who entered university in 1945-46. (The percentage of veterans in each class was 50%, 93%, and 81% respectively). The presence of a few graduates from 30 to 34 years of age was attributed to the men whose university studies were interrupted by the war and to those who might not have taken university training had it not been for the war and the Department of Veterans' Affairs.

It is interesting to note here that the average graduating age of the cross-section considered was approximately 3 years higher than the present-day average. This corresponds very neatly with the 3.3 years of war service put in by the 74 veterans.

Degrees

Of the 93 graduates some 21 have taken post-graduate work at the Master's degree level and of these, 16 took their M.A.Sc., at the University of Toronto. The high percentage (22.6%) of Master's students stems from the fact that nine graduates were sent on advanced degree courses by the R.C.A.F. One-third of the Master's students went on to take their Ph.D's, i.e., approximately 7% of the original graduating class. Twenty of those who have not yet taken their Ph.D. or Master's degree have indicated their desire to do so when the opportunity arises.

Engineering Societies

It was felt that 65 of the graduates who are working in Canada could be considered eligible for membership in the C.A.I. Of these, 42 (62%) are already members. Thirty-six out of an eligible 81 (44.5%) belong to the Institute of the Aeronautical Sciences while only four are members of the Royal Aeronautical Society. These figures would appear to indicate a decided orientation of the Canadian aeronautical engineer to American practice rather than British.

Twenty-two of the graduates are members of the Association of Professional Engineers of Ontario. Among the many other professional groups represented are the Engineering Institute of Canada, The Royal Meteorological Society, the American Society for Quality Control, the American Society of Mechanical Engineers, etc. It was noted that none of those who answered the questionnaire was a member of the Society of Automotive Engineers.

Employment

Table 1 shows the different occupations in which the 93 graduates are now engaged. The greatest percentage (31.2%) have ended up in aircraft construction, 30.1% are working for the government or the armed forces, and 12.9% are in aero-engine construction. Twelve people have never been, or are no longer, associated with the aviation industry and are engaged in such diverse pursuits as general construction, manufacture of welding equipment, public utilities etc. This "etc" masks the curious fact that if you took aeronautical engineering at the University of Toronto during this period you had a better than 2% chance of ending up as a clergyman.

TABLE 1
OCCUPATIONS

Employer	No. People
Aircraft Constructor	29
Canadian Military Services	22
Aero Engine Constructor	12
Repair and Overhaul	6
Government Service	5
Foreign Military Service	2
Airlines	2
Others	15
Total	93

On the questionnaire the graduates were asked to indicate the type or types of work which they are now doing. Since the questionnaire was concerned primarily with aeronautical engineers, the answers received from the twelve individuals no longer in the aviation industry were not considered here.

TABLE 2
EMPLOYMENT FIELDS
(In order of frequency of occurrence)

1. Flight Testing
2. Administration
3. Aerodynamic Analysis
4. Aircraft Maintenance
5. Structural Analysis
6. Policy and Planning
7. Wind Tunnel
8. Aircraft Design
9. Engine Design
10. Aerodynamic Research
11. Accessory Design
12. Production
13. Structural Research
14. Aeroelastic Analysis
15. Structural Testing
16. Sales and Service
17. Air Regulations

— and —

Quality Control, Flying, Public Relations, Flight Safety, etc.

Table 2 lists in order of their frequency of occurrence the fields of employment of the remaining 81. Since several engineers listed more than one such field it is questionable whether the table does more than serve as a guide to types of employment in order of popularity. For example, eighteen people reported that their occupation involved flight testing thus placing this field at the head of the list. If the questionnaire had clearly indicated that only the most important field of activity was to be mentioned it is likely that aerodynamic analysis or possibly administration would have come first on the list.

The graduating classes of 1948-49-50 have had an overall average of 8.7 years experience in aviation, including 5.6 years experience since graduation. The overall figure includes wartime service but not time spent at university in obtaining degrees. Our hypothetical graduate has changed his job once since graduation (more precisely, he has had 1.7 employers). Almost invariably, those in the armed services have had no other employer since graduation and, as a result, the corresponding figure for the civilian engineers is 1.9. The

civilian graduates have worked for their present employers 4.5 years and service graduates over 9 years, the average for all graduates being 5.9 years. Promotions for civilian engineers have averaged 1.8 per man, but for the servicemen the figure is almost exactly 1.

Graduates were asked how long they expect to remain with their present employer. From the members of the Canadian services the reply was an almost unanimous "indefinitely" as might be expected. Of those in industry twelve answered only with a question mark which left 47 returns to be evaluated. Thirty-one of these (66%) indicated their intention to remain where they are indefinitely; five (10.6%) may move at the end of five years; seven (14.9%) will remain where they are for at least two years; and four (8.5%) contemplate a move within one year's time. Thus we see that approximately one third of those considered are thinking of changing their jobs within the next five years.

To finish off the section on employment, the questionnaire contained a query concerning the type of work the engineer hopes to be doing in ten years time. Fifty per cent said that they wanted to continue with some engineering work and 12% wished to be employed as consultants. Five per cent said they would be interested in research and 10% indicated their desire to do at least some teaching. Half of those who replied hoped to be doing some administration along with whatever else their job entails.

Economic Situation

In analysing the questionnaire returns it was soon apparent that more specific information could usefully have been requested. Salary figures were tabulated in $500 increments only, in the hope that a higher percentage of replies would be received; however, the response was so encouraging that we felt more explicit information could have been obtained. Since the salary figures were listed in $500 increments, for the purpose of calculation the mean salary in each range was chosen. It was felt that inaccuracies introduced by such an assumption would not seriously affect the value of the survey.

As mentioned earlier only 77 of the questionnaires were analysed in this section. Those from the individuals no longer engaged in aviation work or whose salary was not paid in dollars were omitted. It was noted that well over half of the 77 are benefiting from some form of pension plan and 46 come under a medical scheme as well. A few engineers also benefit from insurance schemes, bonuses, or overtime. Those in the service who are aircrew or ex-aircrew can collect risk allowance for flying. However, since none of this information was sufficiently explicit to lend itself to evaluation these additional benefits were not included when calculating the average salaries.

Figure 2 shows, in histogram form, the starting and present-day salaries for the three graduating classes surveyed. Table 3 indicates the average of these salaries. Several interesting points emerge from the figure and table:

1. There was a steady increase in starting salaries from 1948-50.
2. In spite of lower starting salaries the 1948 graduates are still better paid than those of 1949-50.

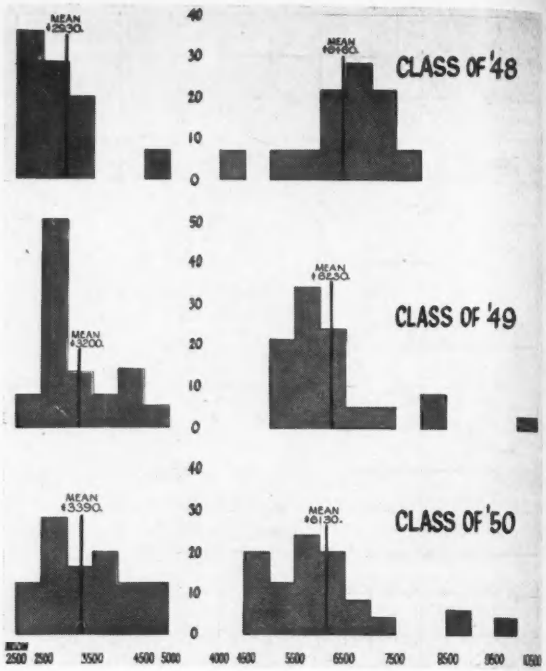


Figure 2
The Salary Picture

TABLE 3
AVERAGE SALARIES
(By year of graduation)

Graduation year	Starting Salary	1955 Salary	Number people
1948	$2930	$6460	14
1949	$3200	$6230	38
1950	$3390	$6130	25

3. The high concentration of starting salaries in the $2,500-3,000 range for the 1949-50 classes caused by the absorption of a large percentage of the graduates by one company.

The results shown in Figure 2 and Table 3 were derived from returns in all 77 questionnaires studied. It was obvious that many factors must have affected each individual's starting and present-day salary, so that an effort was made to analyse these factors to show the effect of each on salaries. Among those considered were degree status, country of employment, military service, age, experience, number of employers, number of promotions, etc., but the only factors in which any correlation was evident were the first three. After this we were forced to conclude that there is no real substitute for ability.

The effect on salary of country of employment appeared to be the most obvious factor, therefore, the graduates were first divided into two categories – those working in Canada, and those working in the United

TABLE 4
AVERAGE SALARIES
(By country of employment)

Country	1955 Salary	Number people
Canada	$5990	67
U.S.A.	$7950	10

States (see Table 4). At first sight it would appear that those employed below the border were almost $2,000 per annum better off than their Canadian counterparts, but these figures had to be taken with a grain of salt for several reasons:

1. Almost all those working in the United States have done post-graduate work at the Ph.D. or M.Sc. level.
2. Although it is known that American salaries are generally higher than Canadian ones, the sample available to us was inadequate for any accurate comparison.
3. Since the number of individuals working in the U.S.A. was relatively small, the overall salary average was boosted considerably by the high salaries of a few individuals.

TABLE 5
AVERAGE SALARIES
(By degree level)

Degree	1955 Salary	Number people
B.A.Sc.	$6030	59
M.Sc. (or equiv.)	$6280	13
Ph.D.	$8550	5

The next step then was to evaluate the effect of post-graduate degrees on salaries. Table 5 shows the average 1955 salaries of the graduates after classification by degree. We were not surprised to find that higher education still brings its own rewards; however, since four of the five Ph.D's work in the United States, the average salary figure is perhaps not a fair one. For purposes of comparison it should be noted that only one of the 13 Master's students is now working in the U.S.A.

A large percentage of the replies received were from graduates now in Canadian military service. These were next sorted out and considered as an independent group. Table 6 shows the average 1955 salary by degree status for this group compared with the remainder (there were no individuals with Ph.D's in the services). It was interesting to note that the larger proportion of those with Master's degrees are employed by the armed forces and that their salaries are slightly below those earned in industry. Comparison of salaries for those resident in Canada showed the overall average for graduates in the RCAF or RCN to be $6,000 and for those in industry to be $5,980.

Finally, the average salary for the Canadian aeronautical engineer was compared with the average for all other Canadian engineers. The basis of comparison

TABLE 6
AVERAGE SALARIES
(Service vs. Civilian)

	B.A.Sc.		M.Sc.	
	Salary	No. People	Salary	No. People
Service	$5860	13	$6190	9
Civilian	$6070	46	$6500	4

was a chart prepared by the Association of Professional Engineers of Ontario. Our hypothetical aeronautical engineer was earning $5,990 and had 5.7 years experience; his counterpart in the Ontario industry was earning approximately $5,750 at the beginning of 1955. It would appear, therefore, that the aeronautical engineer in Canada has little financial advantage over his engineering brethren.

CAUSE FOR COMMENT

In addition to the general requests for information concerning age, salary etc., our questionnaire contained several leading questions designed to provoke comment and personal opinions. In this respect we prompted a much greater response than we had bargained for. Many of those who replied had very definite opinions and a remarkable freedom of expression. Unfortunately this happy state of affairs called for a certain amount of editing, summarizing, and censoring for the written report. The questions which elicited remarks of one sort or another were:

1. Do you think that your specialized undergraduate aeronautical training has given you a definite advantage on the job over other engineers with more general training?
2. Do you think that Canadian aviation offers chances of permanent employment and adequate opportunity for future advancement for the Canadian engineer?
3. Would you (or did you) leave Canada for a better job?

Eighty-seven engineers answered the question concerning the undergraduate training and 59 (63%) thought that their degree work had given them a real advantage. Many who felt that the course was an asset to them suggested changes to make it more valuable to future graduates. The suggested changes were almost evenly divided between requests for more specialization and requests for more fundamentals. Some felt that the course was primarily of value in obtaining their job and that afterwards it was up to the individual to learn for himself. Several of the complaints concerning the course's shortcomings were aimed at lack of training on mechanical, electrical and servo systems.

Some of the comments concerning the course value are listed below:

"Of slight advantage in the RCAF".
"Looking at aerodynamic formulae, even though you may not recognize it, the other fellow thinks you do!"
"Psychological advantage – aeronautical engineers are considered brains".

"Undergraduate work at the University of Toronto is better than at most U.S. colleges – it develops the habit of hard work."

"I would probably take mechanical (engineering) next time."

"Emphasis should be on fundamentals – more time for thinking is required."

After Defence Production, What?

When the answers to the last two questions were considered, we found it difficult to separate the replies since there was definite correlation in many cases. The answers to the third question from individuals now living in the United States definitely reflected their opinions on the second question. Similarly many of the Canadian-employed engineers used their answers to the second question as the basis for their reply to the third. Statistically the replies were as follows:

Question 2—Opportunity for continued employment in Canadian aviation (93 replies)	Yes	69%
	No	19%
	Don't know	12%
Opportunity for future advancement (93 replies)	Yes	63%
	No	25%
	Don't know	12%
Question 3—Would you (or did you) leave Canada for a better job? (83 replies)	Yes	41%
	No	53%
	Don't know	6%

"From the end of the defence production line, where?" This quotation fairly well sums up the comments of those who questioned the future of Canada's aeronautics industry. Another quotation "opportunity in other engineering fields is and will be greater. Unless someone were truly a deep-dyed aviation fanatic, I personally would advise him to try a different field." And again, "the cancellation of just one contract could leave the engineering staff stranded." Other comments in a similar vein indicated the misgivings with which Canadian engineers look at what they referred to as the "instability of the aircraft industry."

Most of the graduates, however, felt that defence spending would be maintained at a level sufficient to keep the industry and its engineers alive.

Many of those who seemed content with their prospects of promotion in the Canadian industry, and most of those who didn't, were unhappy with the attitude of their employers. In fact it was this attitude which was cited by those who left Canada as one of the principal reasons for not remaining in the country.

A few of the comments by Canadian-employed engineers should be quoted here.

"It is high time professional bodies took an active interest in the plight of the engineer in industry – he is swamped in an atmosphere of 'union employeeism' ".

"(There was) recently a great cry of shortage of engineers for aircraft work (but) efficient use of present staff is ignored".

"Conditions are not suitable for the build-up of a stable engineering team (with) a high state of morale."

"There is no interest (by management) in engineering personnel other than as a machine."

"There appears to be no interest in taking engineers through a co-ordinated development programme."

"(There is) no apparent plan in the military services to keep graduate engineers within their fields of training."

"Industry is young as are the engineers employed in it. You work for a young man and chances of advancement are small. Because competition is non-existent you can't advance by moving sideways."

"This is a young man's industry and only expansion brings advancement."

"(There is) little opportunity for promotion of an above average engineer (in the military services)."

When it came to analysing the replies to the question on willingness to take a job outside Canada, we found that ten of the 93 had no opinion or did not care to make a statement. Since there was little likelihood of any of the military service members leaving Canada for greener pastures, their replies were also deleted from the calculations to produce the final results.

Willing to leave Canada	45%
Not willing to leave Canada	47%
Undecided	8%

That is, approximately half of those who are free to make a choice are willing to leave Canada if a better opportunity presents itself.

As might be expected, it was from the engineers employed in the United States that the most pungent comments were received. One capable engineer, now working in the U.S.A., said that he left Canada because Canadian research facilities showed no interest in using his talent. He is extremely bitter towards those whom he feels were responsible, and has taken out American citizenship papers. Some provocative comments from the U.S.A., were:

"An American university offered me the opportunity for complete use of my training – the Canadian industry and research (institutes) did not appear to be aware of the ability of the Canadian engineers, which is held in the highest regard in this country."

"(There is an) apparent lack of interest by Canadian aviation firms and research establishments in hiring those with doctorates except at pay rates 30 to 50% less than corresponding American firms."

"Canada does not seem to appreciate her own professional talent – Canadians do not have to take a back seat to Americans."

This picture must appear more one-sided than is actually the case. The most quotable quotes did not originate with satisfied engineers, in fact such quotes were conspicuous by their absence; however, the questionnaire showed clearly that the majority of the engineers who made no remarks were generally satisfied with conditions as they exist at present.

CONCLUSION

We would have been less than human and more than objective if we had not drawn our own conclusions from the survey results. However, because our opinions are certainly biased, we did not feel they should be given more weight than those of the other 91 who had completed the questionnaire, and they have not been printed here.

Incomplete and inadequate though the survey may have been, we feel it will have served a useful purpose if it provides some food for thought. In the future, perhaps, someone more competent than ourselves will undertake a thorough investigation of the aeronautical engineer and his place in Canadian aviation.

C.A.I. LOG

SECRETARY'S LETTER

ATTENDANCE AT MEETINGS

ATTENDANCE figures are usually given in the Branch Secretaries' monthly reports on their meetings and it has often struck me that these figures are low. I am not familiar with the experience of other societies but it would seem to me that at least 30% of the membership of a Branch should turn out for each meeting – or at least the attendance at each meeting, including guests, should be no less than 30% of the membership. Yet in our Branches the attendance at meetings is frequently below this percentage.

Of course sometimes the reasons have been fairly obvious. The weather has been foul and, once home, people have had all the ice and snow they could take for one day. Or the notice has gone out late and people have made other plans for the evening by the time they have learned of the meeting. But these things do not happen all the time; their occurrence might cause a sudden drop in attendance now and then, but a consistently poor attendance cannot be attributed to these calamities.

Last September some Branches sent out questionnaires to all their members, in an endeavour to find out what programmes the general membership wanted. The response to these questionnaries was quite good and the Programmes Committees went to some pains to comply with the wishes expressed. But despite these efforts, the attendances have been disappointing.

I wonder why. I am just musing out loud; I have no idea of the answer. But I am rather concerned because it is not a healthy state of affairs. Many reasons have been suggested, such as insufficient publicity and unattractive notices, a meeting room with an inappropriate atmosphere, meeting dates conflicting with other activities, meeting times being too early or too late, etc., etc. But, although all these may be contributory reasons, affecting a few members here and there, I do not think that any of them, or even a combination of them, is the real cause of the trouble. Apathy obviously has something to do with it but in a new organization like the C.A.I., apathy has hardly had time to take root.

On the other hand, I must say, the attendance at some Branches has run fairly consistently at over 60% throughout the season. What is their secret?

Perhaps we could start some discussion about this in the Journal. If anyone has any complaints about Branch programmes, or any suggestions how they can be made more attractive, let us air them. Believe me, your Branch Executive and Programmes Committees are anxious to serve their members and want to find the cause of these poor attendances.

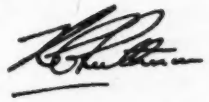
HEADQUARTERS STAFF

At the beginning of February we had a shuffle of our Headquarters Staff, in which Mrs. Ross, who has hitherto been my secretary, was appointed "Assistant to", and Mrs. P. Whipple came in to take her place. Mrs. Ross' chief responsibility will be the Journal and I hope that, when she has relieved me of its tyranny, I shall be able to get around a little more and generally to devote more of my time to Secretarial duties.

MONTREAL

As the first fruits of this arrangement, I managed to get to Montreal for their February meeting and I thoroughly enjoyed it. I took the opportunity to spend an extra day discussing Student matters and the planning of the Annual General Meeting at the beginning of May.

Let us have a good attendance at that.



BRANCHES

EDMONTON

THE following members have been elected to represent the Edmonton Branch on the Council

Mr. G. L. Best and
Mr. C. C. Young.

NEWS

Montreal–Reported by E. Norsworthy

January Meeting

The January 18th meeting of the Montreal Branch heard a discussion by Mr. W. Farr of The Rohm & Haas Co. Bristol, Pennsylvania, on recent developments in acrylic aircraft glazing materials. In introducing the Speaker, Mr. John Auston of Crystal Glass & Plastics drew attention to Mr. Farr's long experience in the aircraft industry, going back to 1926, and mentioned that few aircraft Companies in the United States had not profited from his long experience in the field of acrylic plastics since 1940 when he joined Rohm & Haas, at first being in charge of their development laboratories and later in technical Sales and Service.

Mr. Farr, in his talk, traced the development of acrylic plastics in aircraft from 1938 when 12 x 18 inch sheets were used in the "greenhouses" of the aircraft of that day, through the 1940-1944 period when Plexiglas IA was established as the standard glazing material for wartime aircraft to the development of Plexiglas II for pressurized aircraft in 1944 and the later development of Plexiglas 55 in 1950. He pointed out that Plexiglas 55 possessed high stress crazing resistance and that this material, which had been introduced relatively recently, was the most improved version of Plexiglas currently available in commercial quantities.

Mr. Farr went on to describe the improvements that could be obtained by stretching Plexiglas and described the experimental work in which his Company was at present engaged under a USAF development contract. Experimental work to date had indicated that a marked increase in impact resistance, strength, notch resistance, and resistance to stress and solvent crazing had been obtained when rectangular sheets of Plexiglas had been stretched bi-axially, the only apparent drawback being a lowering of the abrasion resistance of the new material. The properties developed by this new material represent a significant improvement over existing commercially available material and the results obtained from samples being used by various aircraft operators in the United States were awaited with interest.

Mr. Farr was kind enough to answer a great many questions on the effect of temperature, pressure, mechanical damage and cleaning solvent on the canopies of current aircraft. The information he gave was of particular interest since several members present were engaged in an investigation of a series of canopy failures of an aircraft at present in service. Mr. Farr's opinions in this regard were particularly valuable.

At the conclusion of the question period, Mr. Farr was thanked by Mr. C. J. McKyes, President of the Hickey Plastics Company. Approximately 55 members attended the dinner meeting.

February Meeting

The February meeting of the Montreal Branch of the Canadian Aeronautical Institute was held on February 15th, the Speaker being Mr. B. J. Kaganov, AFCAI, Chief Structures Engineer, Canadair Limited, whose subject was "Aeroelasticity". Mr. Kaganov was introduced by Mr. F. Moore of Bristol Aero Engines who drew attention to Mr. Kaganov's wide experience, which included affiliation with Curtis, Brewster, Eastern Airlines and Boeing, prior to taking up his present position with Canadair.

Mr. Kaganov prefaced his talk by mentioning that the field of aeroelasticity was so wide that he would, of necessity, be required to limit himself to a definition of some of the various aeroelastic problems that must be coped with by present day designers. He pointed out that aeroelasticity, which included problems in flutter, aileron and wing divergence, did not become of primary importance until the early stages of World War II as prior to that time, aircraft speeds were low enough so that a structure made to design load requirements was sufficiently rigid. As speeds went up, however, aircraft began to encounter a wide range of problems now defined as aeroelastic. The first of these problems to be seriously considered was flutter, which is usually classed as oscillatory instability in which the flexibility of the structure plays an important part. Mr. Kaganov defined damped, constant and divergent vibrations and went on to define the critical flutter speed as a speed where the system is neutrally stable and the vibrations remain constant. This is based on the fact that at this speed the airplane becomes unmanageable and any increase in speed will cause the vibrations to diverge with consequent catastrophic results. Mr. Kaganov described aileron flutter resulting from mass unbalance of the aileron as being a simple example and went on to consider the sort of modes and frequencies obtained on an actual aircraft. He classified the modes into two broad classes, symmetric and anti-symmetric, describing the characteristics of each class. He described the procedure used to check new aircraft by initial calculation followed by actual checking prior to first flight by direct measurement of the resonances, and detailed the technique used in direct measurement with particular reference to the CL-28.

Mr. Kaganov went on to point out the necessity for structural members to be designed for an airload distribution corresponding to the aircraft structure when deformed and defined the divergent speed, pointing out that the torsional deformations in the swept wing were more favourable in this regard than in the straight wing. The Speaker then described aileron divergence and tail divergence, mentioning that divergence in itself rarely led to the failure of the part by direct action since the secondary effects of a near-divergent condition would probably cause a premature failure. The general cures for divergence were either to increase the torsional stiffness of the wing or to have the flexural axis ahead of the aerodynamics centre. Mr. Kaganov pointed out that divergence and flutter are problems of aeroelastic instability and that at speeds beyond the critical speeds structural failure would occur. He said that this was not true with aileron reversal which could be defined as a condition of no response to applied aileron. He described how aileron divergence occurred and defined the aileron reversal speed as that speed where deflection of aileron would produce no rolling moment. The three critical speeds therefore are flutter speed, divergence speed and aileron reversal speed. On a straight unswept wing of conventional construction, wing torsion-

al divergence usually occurred at a considerably higher speed than aileron reversal which in turn was usually higher than flutter speed. For swept back wings, the aileron reversal speed is lower than the flutter speed and the divergent speed is very high.

Finally, Mr. Kaganov examined the aeroelastic problems introduced by the flexibility of the aircraft structure during landing or upon encountering a gust. After describing these major problems, Mr. Kaganov pointed out that there were numerous other investigations in process, i.e., propeller vibration, engine vibration, control system problems and last but not least, the effect on noise control and sound proofing.

At the conclusion of his talk, Mr. Kaganov showed some German films obtained from the National Research Council, which described flutter experiments carried out in a Full Scale tunnel. Mr. Kaganov was thanked by Mr. Hugh Reid of T.C.A. Approximately 100 members attended the meeting.

Ottawa—Reported by Dr. A. Jaworski

February Meeting

The February 8th meeting of the Ottawa Branch of the Institute was held as usual on Wednesday at 8.30 p.m. at the Beaver Barracks, Argyle Street, and the Chairman of the Branch, Mr. J. L. Orr, presided. It started with an announcement from the Chair that the Nominating Committee was called upon by the Executive to submit a list of candidates for the officers during the 1956-57 season.

The Chairman underlined that the present meeting should be called a Navy meeting, and surely it was. Capt. C. G. Daniel, in introducing the speakers, Commander (P) H. J. Hunter, Deputy Director of Naval Aviation, R.C.N., and Commander (E) J. R. Llewellyn, Engineer-in-Chief's Department, R.C.N., pointed out that the subject "Naval Aircraft Carrier Operation" would be presented in three parts. In part one, a film was to be shown (U.S.S. Antietam, Angle Deck). The film was obtained on loan from the U.S. Navy and was flown on Monday from Washington especially for the meeting. In part two, Commander Hunter would describe the flying operation, and finally in part three, the engineering side of the deck activities would be presented by Commander Llewellyn under the title "Flight Deck Machinery in Aircraft Carriers".

The film ran for about twenty-five minutes and some of the details of the flying operation were shown in slow motion at one-quarter of the actual speed. The presentation was quite realistic when a picture was seen of an aircraft which, after landing on a deck, crashed into some aircraft standing in a parking area.

Commander Hunter presented the flying operation as a pilot would see it during an actual hunt for a submarine, from the moment when the pilots are called for briefing until the mission is completed. A pilot's experience for such an undertaking requires at least 1,500 flying hours, 600 of them on jets. Usually when a mission is successfully completed, there is fuel left for no more than twenty minutes flying time, to cover any emergency.

The more recent technical improvements, such as the angle deck, steam catapults, and a reflection mirror which indicates a correct landing path, were taken into account in the description of the flight operation. It may be noted that the back-history of the reflection mirror was even more interesting than Newton's famous apple, as the application of a mirror for a landing aid started in an office of the British Admiralty where an enterprising gentleman played with his secretary's compact.

Summing up, the necessary characteristics of an aircraft suitable for carrier operations were briefly sketched; the visibility aspect was strongly underlined, the reliability of all instruments in general, and a horizon indicator and fire warning devices in particular. (The first is unreliable immediately after a sharp turn, and the latter very often frightens the pilot with a false alarm.) A new dial for an altimeter which will eliminate any ambiguity in readings, e.g. 11,000 for 1,100, is highly desirable. A plea was made for an easily handled mechanism to eject the canopy during emergency.

The third part, "Flight Deck Machinery in Aircraft Carriers" was illustrated by Commander Llewellyn with several drawings. The general principles of constructing arresting gear came under three headings; (a) a smooth build-up of absorbed energy, (b) a sustained period, and (c) a retardation movement. In the shock absorbers, the travelling ratio of 12 : 1 is commonly used, whereas for the barriers, it is of an order of 6:1.

It was pointed out by the speaker that the development of catapults preceded the development of arresting gear, as the requirement for the former was imposed by the use of reconnaissance aircraft from cruisers and battleships whereas, in early days, the low speed of the aircraft presented the problem of catching up with the ship; an arresting gear was quite unnecessary! The launching medium in catapults has changed with time – cordite to be replaced by compressed air, and more recently by steam – but there are already indications that some other medium may replace steam too. The difficulties facing the constructor of a catapult were reviewed; he must build a mechanical system that must stand up to acceleration of 10 g, with a speed of 70 knots and be brought to rest suddenly, all within a time cycle of an order of 70 seconds. Some valuable lessons about the use of slot cylinders for the catapults had been obtained from inspection of the German launching chutes for the V-1's.

After a lively discussion period, the meeting ended at 11.00 p.m. To judge from the favourable comments indicated on the attendance cards, the 50 members and guests who attended evidently enjoyed the meeting.

Vancouver—Reported by R. J. McWilliams

February Meeting

The meeting was held at the Aero Club of B.C., at 20.00 hours. It was preceded by a short business session lasting twenty minutes and was attended by seventy-one members and guests.

The speaker for the evening, Mr. A. L. Fornoff, Manager, Aircraft Sales, Bell Aircraft Company, was introduced by a man whose name is becoming a by-word in Canadian Helicopter operations – Mr. Carl Agar, of Okanagan Helicopters Limited.

Mr. Fornoff opened his subject "Helicopter Utilisation" by speaking of the inertia of public acceptance. The answer to the problem – knowledge; the tool – publicity. He spoke at some length of the archaic ordinances restricting helicopter operation in built-up areas, and went on to discuss the various features which should be stressed in the crusade to overcome resistance.

The helicopter was compared with conventional fixed wing aircraft, and with the convertiplane and V.T.O. types. The various fields in which the helicopter is considered to be outstanding were reviewed and its adaptability emphasized. This adaptability was excellently illustrated with the use of a recently produced film.

Mr. Fornoff closed his talk by referring to the latest in the Bell 47 series, the Model 47J. The various features designed to improve customer appeal were illustrated by means of coloured slides.

The talk was followed by a short question and answer period. Our Speaker was thanked by the Secretary and the meeting adjourned at 22.00 hours.

MEETINGS

ANNUAL GENERAL MEETING

An outline of the programme for the Annual General Meeting of the Institute appears below. The complete programme is set out in a Notice which is being mailed to all members.

Hotel Reservations

A block of rooms has been set aside in the Sheraton-Mount Royal Hotel for those attending the meeting. These rooms will not be held after the 25th April but this does not mean that there will be rooms available until that date; if all the rooms are taken by, say, the 18th April, anyone trying to reserve a room on the 20th will be disappointed. It can happen. It happened last November in Ottawa. Members are therefore advised to make their reservations as soon as possible.

Dinner Reservations

It would be very helpful to the Committee if members planning to attend the Dinner on the 3rd May, would buy their tickets early. Dinner Reservation Cards are being distributed with the Notices; they should be completed and sent in to Mr. W. R. Cuff, Box 141, Station O, Ville St. Laurent, Montreal 9, P.Q. as soon as possible and not later than the 30th April.

It is a very bad habit to "get your Dinner Ticket when you get there." It makes planning very difficult.

Reserved Tables

There will be no reserved tables, except for the ladies of those sitting at the head table and for the Press.

Exhibits

Sustaining Members who wish to take advantage of the Meeting to arrange exhibits of their products in private rooms in the Sheraton-Mount Royal are asked to make their own arrangements with the Hotel. A bulletin board will be provided close to the Registration Desk on which notices of such exhibits can be displayed.

ANNUAL GENERAL MEETING

SHERATON-MOUNT ROYAL HOTEL
MONTREAL

3rd and 4th MAY, 1956

May 3rd	Morning—9.00 a.m. to 10.30 a.m.	Business Meeting	
	Morning—10.45 a.m. to 12.45 p.m.	Propulsion	
	Afternoon—2.00 p.m. to 5.00 p.m.	Manufacturing	
	Evening—7.00 p.m.	Dinner	
May 4th	Morning—9.00 a.m. to 12.00 noon	Design	
	Afternoon—2.00 p.m. to 5.00 p.m.	Aerodynamics	Concurrently
		Maintenance and Operation	Concurrently

The Principal Speaker at the Dinner will be

AIR VICE MARSHAL M. M. HENDRICK

Air Member Technical Services

The McCurdy Award for 1955 will be presented at the Dinner by

THE HONOURABLE J. A. D. McCURDY

SUSTAINING MEMBERS

NEW SUSTAINING MEMBER

THE following Company has joined the Institute as a Sustaining Member:

Dorval Metalcraft Company

NEWS

The Goodyear Tire and Rubber Company of Canada Ltd. announces the development of a watermelon-shaped rubber tank that could revolutionize the bulk transportation and storage of fuels and other liquids.

Called the Rolli-Tanker, the container can be rolled over ground, floated in water and dropped without bursting. It is expected to meet the needs of manufacturers, builders, farmers, the armed forces and others faced with liquid handling problems.

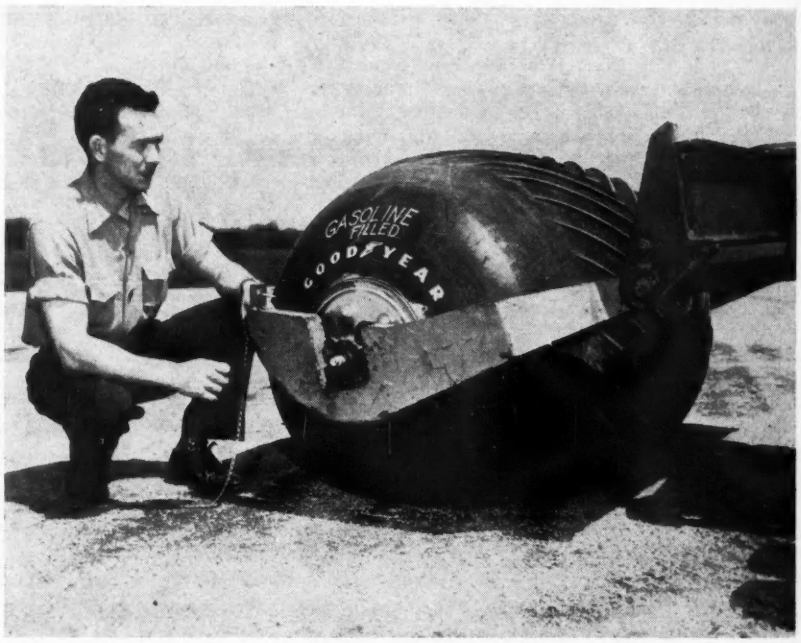
Rolli-Tankers are unconventionally-shaped tires of nylon cord and tread stock construction with fuel-proof inner lining. They can be built in a range of sizes. To date the company has tested 3½ x 5-foot tanks that weigh 40 pounds deflated and give the appearance of oversized watermelons when loaded to 250-gallon capacity.

Mounted on hubs and axles to permit easy handling, Rolli-Tankers may be towed manually or by vehicle. The containers have excellent floatation characteristics because of extremely low ground-bearing pressure. Only 30 pounds of drawbar pull—easily supplied by one man—are needed to roll the storage units.

Filled Rolli-Tankers can be towed either singly or in tandem arrangements behind any vehicle with a trailer hitch without affecting the vehicle's gross weight. Low footprint pressures permit them to be easily drawn over unprepared surfaces, such as rocky, uneven terrain, swampy soil and sandy areas. The pliable construction of the tankers conform to the shape of obstacles over which they pass.

Rolli-Tankers have a natural buoyancy, too, and can be floated down rivers and across lakes when filled properly.

During recent tests conducted by Goodyear's Aviation Products Division, fully loaded tanks were dropped 15 feet without bursting, giving preliminary indication that Rolli-Tankers may be suitable for parachute or free fall droppable containers.



The Goodyear Rolli-Tanker

D. Napier and Son Ltd. announces that the Oryx, to the N.Or.1 rating, has now successfully completed a 150 hour type approval test to the requirements of the British Ministry of Supply turbine engine type schedule. This is the first British gas turbine engine to pass a type test for helicopter application. The engine has been given type approval at a maximum rating of 780 gas horsepower with a recommended initial overhaul life of 250 hours.

Aviquipo of Canada Ltd. announces that they have been appointed exclusive distributor for the Champion Aircraft Corporation of St. Paul, Minn., for Ontario, Quebec and the Maritimes. The Champion 7-EC, a two-passenger, utility aircraft with Continental C90-12F 95 hp engine is a type likely to prove very useful for farmers, ranchers and others; it is a successor to the well-known Aeronca Champion. Aviquipo is now seeking the appointment of dealers.

They have also been appointed the Canadian representative of Manning, Maxwell & Moore Inc. This company manufactures Turbojet Engine temperature control Amplifiers, Electronic Amplifiers, Pressure Switches for Pressure Gauges, Thermocouples, Hydraulic Valves and Jet Engine Afterburner Control Systems.

Computing Devices of Canada Ltd. has located Canada's first Data Processing Centre at its head office in Ottawa. This Centre is available to provide government and industry with such services as require computer facilities, both digital and analogue, for problem formulation and analysis, programming, computation, data reduction, application of data processing equipment for business procedures and practical instruction in computer operation and programming.

PSC Applied Research Ltd. have issued some details of their Type T246 Mk 2 Automatic Tri-Film Processor. This is a portable equipment designed to process and dry automatically 16 mm, 35 mm, and 70 mm film.

Particular attention has been given to the facility of processing various sizes of film and simplification of loading procedure. Four lengths of 16 mm film or two 35 mm lengths can be handled simultaneously, or one 70 mm length. The various film sizes are accommodated by simple adjustments of film separators. The film magazine holds four 400 ft

16 mm films, two 400 ft 35 mm films or one 400 ft 70 mm film, spooled or on cores. Larger magazines can be provided on special order. Three basic speeds of 1½, 3 or 6 ft per minute can be obtained and are changed by a simple belt adjustment. Much higher speeds in the same ratios are also possible. Further fine control over processing is obtained by a special programming assembly which can be adjusted to change the loop length in each individual processing tank. The film is drawn from a light-tight magazine, formed into loops in the successive process tanks, transported at pre-selected speed and dried in a forced hot air chamber, after which it is wound on take-up spools. Tendency-driven take-up shafts are used to assure uniform spooling. No lower rollers are used, as the upper roller system of the equipment automatically forms loops of the correct length in each tank. Processing is controlled by a mechanical program unit after the film is loaded into the machine – no special "leader" or continuous tapes, chains or sprockets are used.

As normally supplied, the first two tanks are used for developing, the second pair for fixing and the last two for washing. However, many combinations of tanks and processing solutions, as for stabilization, special hardening or combined development-fixation, may be obtained by changing Saran conductors on the liquid distribution system. Positive-drive diaphragm-heater pumps are installed on one or more of the first four tanks, according to customer requirements. Running water must be provided for the wash tanks. Separate temperature control of the processing solutions is possible on each tank over the range 60° to 110°F., within ± 1°F. Temperature control is exercised in the diaphragm heater pumps, ensuring positive action on the circulating fluid. The need for stop baths and interbath rinses, normally required in many processes, is virtually eliminated because of the positive squeegee roller design. The rollers cause surface liquid to run back into each tank and thus a minimum of solution carry-over occurs from developer and fixer tanks. A high-efficiency blower system and electrical heating ensure rapid drying in the machine.

The latest high temperature chemical resistant plastics and type 316 stainless steel are used in all areas likely to be contaminated by chemicals. The Mark 2 processor is equipped with wheels and handles, and is designed to be moved readily from one location to another. At the same time attention has been given to the provision of easy access to those areas requiring cleaning and periodical maintenance.

MEMBERS

ANNUAL DUES

Dues bills are now being sent out and, since they invariably give rise to a few questions about the period covered and such matters, it may be opportune to give some explanation. It is all laid down in the By-laws but, in their present form, the By-laws are not easy to follow: as mentioned in last month's Journal, an amendment is under way with the object of making them more readily intelligible.

The Institute's year ends on the 31st March (Article 14, Section 2) and each year's annual dues cover the period beginning on the 1st April and ending on the 31st March following. The year which was called "1955-56" extends from the 1st April 1955 to the 31st March 1956. Anyone admitted during the course of the year 1955-56 was liable to pay dues for that year, except in certain circumstances which will be explained below.

Dues are payable in advance and the bills sent out in March 1956 refer to the year 1956-57, that is the year beginning on the 1st April 1956 and ending on the 31st March 1957. The dues for 1956-57 ought to be paid by the 1st April 1956 but, under the terms of Article 10, Section 6, a member is given 6 months grace –until the 1st October–before he becomes technically "in arrears", with the disqualification from voting and other penalties which this technicality entails.

As stated above, anyone admitted during a fiscal year is liable to pay dues for that year. However, the second part of Article 10, Section 5, means that if anyone is admitted after the 1st January–say on the 10th January 1956 for the sake of example–his first dues cover "the unexpired portion of the year of election" i.e., from the 10th January 1956 to the 31st March 1956, and "the fiscal year following", i.e., from the 1st April 1956 to the 31st March 1957; in other words the unexpired portion of the year 1955-56 is thrown in free with the dues for 1956-57. This bonus is given only for the last three months of any year, January, February and March; the line could have been drawn after the 1st February, or after the 1st December, or anywhere, but it had to be drawn somewhere and the 1st January was the place chosen. Anyone who is admitted in December is perhaps out of luck, but it should be remembered that he is put on the mailing list and receives the full services and privileges of membership from the date of his application, which is probably two or even three months earlier–and very little activity takes place before September in any case.

NEWS

W/C E. P. Bridgland, A.F.C.A.I., has returned from his posting in England and has been appointed Commanding Officer of No. 12 T.S.U., R.C.A.F., Toronto.

I. M. Hamer, A.F.C.A.I., has recently been appointed Vice-President and General Manager and continues as a Director of Dowty Equipment of Canada Ltd. Mr. Hamer was a member of the Steering Committee and Interim Council of the Institute and was the first Chairman of the Toronto Branch.

E. Kosko, A.F.C.A.I., Avro Aircraft Ltd., has been appointed Scientific Consultant to the Institute of Aerophysics, University of Toronto. He is conducting on a part-time basis a lecture and laboratory course in Analysis of Aircraft Structures as part of the Fourth Year Aeronautical Engineering curriculum at the University.

W. Parish, M.C.A.I., after some 16 years service with the various companies which have operated the Malton plant, has resigned as Quality Control and Inspection Manager of Avro Aircraft Ltd.

R. Finney, Technical Member, formerly with Railway & Power Engineering Corp. Ltd., has joined Jarry Hydraulics and has been appointed Sales and Engineering Liaison representative of the latter firm in the Toronto area.

STUDENTS

The attention of those Student members who will graduate within the next few months is directed to a recent ruling of the Council concerning regrading.

It should be borne in mind that, according to the By-laws, a Student member is defined as one "undergoing a course of study in a school of engineering or technology recognized by the Council". After graduation from that school he normally cannot qualify any longer for the Student grade.

It has been decided therefore that a Student member must apply for regrading within the fiscal year in which he graduates (the fiscal year runs from the 1st April to the 31st March); in other words, after graduation he has until the following 31st March in which to apply for regrading and, if he has not done so by that date, his membership must be terminated.

There are of course certain exceptions. If a Student after graduation continues a full time course for his Masters degree, he can retain the grade of Student. But he cannot retain the grade if his normal occupation is as a paid employee of a company or organization, even though he may devote a great deal of his out-of-working hours time to post-graduate academic studies. Each doubtful case will be considered by the Council on its merits and if any Student member thinks that his circumstances are such that he still qualifies for that grade, he should submit his case for consideration.

If not, he should apply for advancement in grade within the fiscal year.

ADMISSIONS

At a meeting of the Council, held on the 10th February, 1956, the following were admitted to the grades of membership shown.

Associate Fellow

Dr. E. R. Sharp, Director, N.A.C.A., Lewis Flight Propulsion Laboratory, 21000 Brookpark Road, Cleveland, 11, Ohio, U.S.A.

Member

H. Aass, Aeronautical Engineer, Spartan Air Services: *P.O. Box 166, Manotick, Ont.*

J. P. Beauregard, Performance Engineer, Orenda Engines Limited: *161 Elizabeth St. South, Brampton, Ont.*

R. C. Belcher, Chief Inspector, Standard Aero Engine Ltd.: *184 Kirkfield Park Road, Kirkfield Park, Man.*

F/L E. Charters, Senior Technical Officer, R.C.A.F. Rockcliffe: *30 Arcturus, P.O. Box 403, R.C.A.F. Stn., Rockcliffe, Ottawa, Ont.*

A. T. L. Dyne, Plant Manager, Standard Aero Engine Ltd.: *498 Stiles St., Winnipeg, Man.*

S/L T. V. Gilbert, R.C.A.F. Rockcliffe: *Box 87, R.C.A.F. Stn., Rockcliffe, Ottawa, Ont.*

J. L. Gould, Defence Scientific Service Officer, Defence Research Board, N.D.H.Q. Elgin St., Ottawa, Ont.

H. W. Grant, Plant Engineer, Standard Aero Engine Ltd.: *388 Davidson St., Winnipeg 12, Man.*

F/L J. Harrison, Technical Officer, R.C.A.F.: *361½ Daly Ave., Ottawa, Ont.*

W. F. J. Harwood, Western Canada Service Supervisor, Bristol Aircraft (Western) Ltd., Stevenson Field, P.O. Box 874, Winnipeg, Man.

S. M. Howes, Mechanical Engineer (A), Orenda Engines Ltd.: *c/o C. A. Grinyer, R.R. No. 1, Caledon, Ont.*

A. E. Johnston, Project Coordinator, Avro Aircraft Ltd.: *27 Barrhead Crescent, Rexdale P.O., Ont.*

Z. E. Laviolette, Technical Representative, Canadair Ltd.: *361 St. Eustache St., St. Eustache Co., Deux Montagnes, P.Q.*

W. L. McDonald, Orenda Projects Supervisor, Orenda Engines Ltd.: *49 Fairmeadow Ave., Willowdale, Ont.*

E. H. Monerieff, President, Standard Aero Engine Ltd. P.O. Box 764, Winnipeg 1, Man.

S/L J. E. Neelin, R.C.A.F. Rockcliffe, Ont.: *Box 29, R.C.A.F. Stn., Rockcliffe, Ottawa 2, Ont*

K. J. Orlik-Ruckemann, Asst. Research Officer, High Speed Aerodynamics, National Research Council, Ottawa, Ont.

C. G. Peters, A/C Maintenance Engineer, Okanagan Helicopters Ltd.: *556 West 63rd Ave., Vancouver, 14, B.C.*

R. Salmon, Stress Analysis, Canadair Ltd.: *Apt. 18, 1585 Ouimet St., Ville St. Laurent, P.Q.*

F. R. Saunders, Manager, Field Service Engineering, Sperry Gyroscope Co. of Canada: *95 Angell Ave., Beaurepaire, P.Q.*

J. A. Scholefield, S & S Engineer, Aviation Electric Ltd.: *128 -18th Ave., St. Eustache-Sur-Le-Lac, P.Q.*

G/C A. B. Searle, Commanding Officer, C.E.P.E., R.C.A.F. Stn., Rockcliffe: *20 Carlyle Ave., Ottawa, Ont.*

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F/L K. F. Shepard, Project Engineer, Canuck A/C R.C.A.F., A.F.H.Q.: *690 Cole Ave., Ottawa, Ont.*

D. C. Wallis, Aerodynamicist 'A', Avro Aircraft Ltd.: *Box 296, Woodbridge, Ont.*

Technical Member

F/O L. F. Bateman, Engineering Officer, A.M.C.H.Q., R.C.A.F., Stn., Rockcliffe: *14 Arcturus Ave., R.C.A.F. Stn., Rockcliffe, Ottawa, Ont.*

F/L W. K. Bell, Aeronautical Engineer, R.C.A.F.: *594 Sparksdale Ave., Ottawa 2, Ont.*

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J. K. Finer, Draughtsman "A", Canadair Ltd.: *231 Willowdale Rd., P.O. Box 504, Rosemere, P.Q.*

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R. A. Keyes, Engineer "C", Canadair Ltd.: *Apt. 6, 4655 Dudemaine St., Montreal 9, P.Q.*

MEMBERSHIP OF THE C.A.I.

as at the Meeting of the Council on the 10th February, 1956.

Technical	**1400**
Associates	**60**
Total	**1460**

The Technical grades comprise the following:

Honorary Fellows	10
Fellows	18
Associate Fellows	175
Members	698
Technical Members	385
Technicians	33
Students	80

Technical Member (Cont.)

R. A. Knowles, Field Service Representative, Orenda Engines Ltd.: *325 Fifth Ave., Ottawa, Ont.*

R. E. Langley, Manager, Aircraft Engineering Div. of Standard-Modern Tool Co. Ltd.: *76 Densley Ave., Toronto 15, Ont.*

J. E. Lloyd, Engineer "B", Canadair Ltd.: *4915 Maplewood Ave., Apt. 12, Montreal, P.Q.*

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G/C W. M. Murray, Supply Officer, R.C.A.F., A.M.C.H.Q., Room B, 263, 70 Lyon St., Ottawa.

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Technician

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J. D. Lawson, Univ. of Toronto, Toronto, Ont.

E. P. Muntz, Univ. of Toronto: *583 Huron St., Toronto, Ont.*

Associate

R. A. J. Murison, President, Canadian Flight Equipment Ltd.: *264 D'Arcy St. N., Cobourg, Ont.*

W. F. Murphy, Asst. Director (Contracts), A/C Branch, D.D.P., Ottawa, Ont.

J. F. Taylor, General Manager, Bendix-Eclipse of Canada Ltd., 2444 Bloor St. W., Toronto 9, Ont.

R. M. Wilson, Asst. to Manager of A/C Sales, Canadair Ltd.: *Apt. 201, 3235 Ridgewood Ave., Montreal, P.Q.*

AMENDMENT OF THE BY-LAWS

Ballot papers in connection with the proposed amendments to the By-laws were sent to all voting members on the 28th and 29th February 1956

DO NOT FAIL TO RETURN YOUR COMPLETED BALLOT

by the

23rd MARCH 1956

Votes received after this date will not be counted